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RTO-MP-25

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION

BP 25, 7 RUE ANCELLE, F-92201 NEUILLY-SUR-SEINE CEDEX, FRANCE

RTO MEETING PROCEEDINGS 25

New Metallic Materials for the Structure of Aging Aircraft

(les Nouveaux Matériaux métalliques pour les structures des aéronefs d'ancienne génération)

Papers presented at the Applied Vehicle Technology Panel (AVT) Workshop, held in Corfu, Greece, 19-20 April 1999.

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The Research and Technology Organization (RTO) of NATO

RTO is the single focus in NATO for Defence Research and Technology activities. Its mission is to conduct and promote cooperative research and information exchange. The objective is to support the development and effective use of national defence research and technology and to meet the military needs of the Alliance, to maintain a technological lead, and to provide advice to NATO and national decision makers. The RTO performs its mission with the support of an extensive network of national experts. It also ensures effective coordination with other NATO bodies involved in R&T activities.

RTO reports both to the Military Committee of NATO and to the Conference of National Armament Directors. It comprises a Research and Technology Board (RTB) as the highest level of national representation and the Research and Technology Agency (RTA), a dedicated staff with its headquarters in Neuilly, near Paris, France. In order to facilitate contacts with the military users and other NATO activities, a small part of the RTA staff is located in NATO Headquarters in Brussels. The Brussels staff also coordinates RTO's cooperation with nations in Middle and Eastern Europe, to which RTO attaches particular importance especially as working together in the field of research is one of the more promising areas of initial cooperation.

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- SAS Studies, Analysis and Simulation
- SCI Systems Concepts and Integration
- SET Sensors and Electronics Technology
- IST Information Systems Technology
- AVT Applied Vehicle Technology
- HFM Human Factors and Medicine
- MSG Modelling and Simulation

These Panels are made up of national representatives as well as generally recognised 'world class' scientists. The Panels also provide a communication link to military users and other NATO bodies. RTO's scientific and technological work is carried out by Technical Teams, created for specific activities and with a specific duration. Such Technical Teams can organise workshops, symposia, field trials, lecture series and training courses. An important function of these Technical Teams is to ensure the continuity of the expert networks.

RTO builds upon earlier cooperation in defence research and technology as set-up under the Advisory Group for Aerospace Research and Development (AGARD) and the Defence Research Group (DRG). AGARD and the DRG share common roots in that they were both established at the initiative of Dr Theodore von Kármán, a leading aerospace scientist, who early on recognised the importance of scientific support for the Allied Armed Forces. RTO is capitalising on these common roots in order to provide the Alliance and the NATO nations with a strong scientific and technological basis that will guarantee a solid base for the future.

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New Metallic Materials for the Structure of Aging Aircraft

(RTO MP-25)

Executive Summary

Military hardware is very demanding on structural performance. Advanced materials and processing methods are providing new options for materials selection and component design. The papers presented showed how new metallic alloys and composite materials can improve the performance and life of airframe structures. Larger units can be manufactured by improved processes such as investment casting, welding, creep-age forming and high-speed machining. This reduces the parts count and reduces assembly costs. Life can be extended by new corrosion resistant coatings, but these must be environmentally friendly.

The round table discussion closing the meeting focussed on three main issues requiring further work: integrating sub-structures into larger structures, cost-benefit issues of applying new materials and cost reduction through "Qualification by Analysis".

Larger structures were discussed under the aspects of inspectability and damage tolerance. They may be easier to inspect since they have less fasteners, less holes etc. However, large unit constructions do not contain the crack stoppers inherent in smaller riveted components. This must be considered carefully in design. Also the load path during failure will be different in these new units so substitution needs careful stress analysis and full understanding of the failure modes.

However, these large parts will undoubtedly require coatings to protect them from corrosion and care should be taken to ensure that this does not impair inspectability. Discussion raised the point that civil aircraft experience more corrosion damage, although they are usually manufactured from similar materials. For example, Dassault Aviation see more corrosion problems in business jets than in the Mirage, but business jets fly about twice as many hours per year. British and Canadian experience has shown a high incidence of corrosion in transports and maritime patrol aircraft, although this is often linked to poor design and coating technology which reflects the age of the systems.

The balance of cost against benefit is an important issue. With the small fleet sizes of European airforces it is difficult to introduce new materials to overcome ageing problems. The cost of re-certification testing is too high for the potential benefit. Substitution only becomes an option when there is a serious problem which would require part replacement several times through the life cycle of the aircraft. Since this represents a significant barrier to the application of new materials, the options for removing this barrier were discussed.

A change to a system of "Qualification by Analysis" would provide the step change necessary for materials substitution. This topic has already been raised as a potential AVT Technical Team activity under ageing systems. Certifying authorities insist on full structural tests for new flight-critical components. However there is already some shift away from testing since new versions of aircraft can be approved if they are extrapolations of existing designs and have appropriate laboratory and analytical data to support the new design. Numerical stress analysis of designs is now relatively easy and accurate. However there is still poor quantitative understanding of the effects of ageing and the definition of failure. Qualification by analysis will need a full identification of all the possible failure modes and a risk assessment which quantifies the most important areas of concern. These may change with time as the aircraft role is changed and if damage, such as corrosion or inelastic strains start to affect the load paths in the structure. These are issues which should be addressed in the life cycle management of the system.

les Nouveaux Matériaux métalliques pour les structures des aéronefs d'ancienne génération

(RTO MP-25)

Synthèse

Les performances structurales du matériel militaire doivent être exceptionnelles. Or, les nouvelles méthodes de traitement et les matériaux avancés offrent de nouvelles opportunités pour le choix des matériaux et la conception des composants. Les communications présentées ont montré les possibilités des nouveaux alliages métalliques et des matériaux composites pour l'amélioration des performances et du cycle de vie des cellules. Des éléments plus grands peuvent désormais être fabriqués par le biais de procédés avancés tels que le moulage de précision, le soudage, le formage qui anticipe les effets du fluage et du vieillissement, et l'usinage ultra-rapide. Ces procédés permettent de réduire le nombre de pièces et de diminuer les coûts de montage. Les nouveaux revêtements résistant à la corrosion permettent de prolonger la durée de vie, mais doivent être sans danger pour l'environnement.

La table ronde qui a clôturé la réunion a porté essentiellement sur trois grands domaines pour lesquels il y a lieu d'entreprendre des travaux supplémentaires, à savoir : l'intégration des sous-structures dans des structures plus grandes, la comparaison coûts-avantages de la mise en oeuvre de nouveaux matériaux et la réduction des coûts par « la qualification par l'analyse ».

Les éléments de grande dimension ont été examinés du point de vue de leur contrôlabilité et leur tolérance à l'endommagement. S'ils sont plus faciles à contrôler (absence de fixations, moins d'ouvertures etc...), les grands éléments n'ont pas les arrêts de crique habituellement intégrés aux composants rivés plus petits. Il faut en tenir compte lors de la conception. De même, les voies de contrainte en cas de défaillance seront différentes pour les nouveaux ensembles ; par conséquent tout remplacement doit être précédé d'une analyse poussée des contraintes et les modes de défaillance doivent être bien appréciées.

Cependant, ces grands ensembles devront certainement être revêtus d'une protection afin de les protéger contre la corrosion, et cela ne doit pas compromettre leur contrôlabilité. Il est ressorti des discussions que les aéronefs civils subissent plus d'endommagements dûs à la corrosion que les avions militaires, bien qu'en général, ils soient fabriqués à partir de matériaux analogues. A titre d'exemple, la compagnie Dassault Aviation a relevé plus de cas de corrosion sur les avions d'affaires à réaction que sur les Mirages, même si les avions d'affaires effectuent deux fois plus d'heures de vol par an que les Mirages. L'expérience des britanniques et des canadiens montre que la corrosion se produit fréquemment sur les aéronefs de transport et de patrouille maritime, même si ce phénomène est souvent lié à des défauts de conception et à des technologies de revêtement qui témoignent de l'âge des systèmes.

Le compromis entre coûts et avantages demeure une question importante. Etant donné les flottes relativement réduites des armées de l'air européennes, il est difficile, pour résoudre les problèmes de vieillissement, de mettre en place des nouveaux matériaux. En effet, les coûts des essais en vue d'une nouvelle certification sont trop élevés par rapport aux avantages possibles. Le remplacement ne peut être proposé qu'en cas de problèmes importants, nécessitant le remplacement de pièces plusieurs fois pendant le cycle de vie d'un aéronef. Etant donné que celà représente un obstacle sérieux à la mise en oeuvre des nouveaux matériaux, les différentes possibilités d'élimination de cet obstacle ont été abordées.

L'adoption d'un système de « Qualification par l'analyse » serait un premier pas vers le nécessaire remplacement des matériaux. Ce sujet a déjà été proposé pour une éventuelle activité d'une équipe technique AVT dans le cadre des systèmes vieillissants. Les autorités de certification imposent des essais structuraux complets pour tout nouveau composant indispensable à la sécurité. Cependant, les fabricants font de moins en moins appel aux essais, puisque de nouvelles configurations peuvent être approuvées dans la mesure où il s'agit d'extrapolations d'études déjà réalisées et que la nouvelle conception soit étayée par des données analytiques et des résultats d'essais en laboratoire. L'analyse numérique des contraintes est aujourd'hui précise et simple à réaliser. Cependant, il n'existe que peu de données quantitatives sur les effets du vieillissement et la définition de la défaillance. La mise en oeuvre généralisée de la qualification par analyse passe par l'identification détaillée de toutes les modes de défaillance possibles et une évaluation des risques qui quantifie les domaines les plus préoccupants. Ces domaines risquent de changer avec le temps, au fur et à mesure que le rôle de l'aéronef évolue, et en particulier si l'endommagement, par la corrosion ou les contraintes inélastiques, commence à modifier les voies de contrainte dans la structure. Toutes ces questions sont à aborder dans le cadre de l'examen de la gestion du cycle de vie d'un système.

Contents

* Paper was prepared in time for the meeting but was not presented.

* Paper was available at the time of the meeting but not presented.

Theme

Structural components of aging aircraft can be replaced with components manufactured from materials with specifications of a higher qualification, thus enhancing various parameters including overall life cycle cost (LCC). Recent research has led to development of several new materials, heat treatments and processing technology which can be used for the replacement of components prone to corrosion, stress corrosion and fatigue. Specific examples include: a new T77 heat treatment for 7150 alloy; a new alloy 7055, and new processing to control composition and microstructure, e.g. 2524. New alloys have been developed, e.g.: high strength aluminium lithium alloy 2195; fatigue resistant aluminium lithium alloy 2097; laminated hybrids of aluminium and titanium alloys; Timetal 1100 for up to 1100-F; Ti62222 alloy; lower oxygen versions of Ti-6Al-4V; new beta titanium alloys; improved metal matrix composites; high strength, high corrosion resistant steels; improved Ni-Co alloys and low carbon steels.

These new materials and processes may add significant life to aging aircraft that form the backbone of the NATO operational force structure: the KC-135, introduced more than 40 years ago; e.g. the F-15 air superiority fighter, operational 20 to 25 years ago; the F-16 KC-10 became operational at least 15 years ago. Many are expected to remain in service an additional 25 years or more. Maintaining them is based on economic and safety considerations. Retrofitting damaged components with new advanced materials can considerably reduce life cycle cost. NATO must consider the use of these new materials and the proposed workshop provides a unique forum to discuss and further promote common approaches necessary for keeping international fleets.

Thème

Les composants structuraux des aéronefs en service depuis longtemps peuvent être remplacés par des composants fabriqués à partir de matériaux avec de meilleures spécifications, en vue d'améliorer certains paramètres tels que les coûts globaux de possession (LCC). Les travaux de recherche récents ont conduit au développement de matériaux, de traitements thermiques et de technologies de transformation, dont la mise en oeuvre permettrait de remplacer les composants sujets à la corrosion, à la corrosion sous contrainte, et à la fatigue. Parmi d'autres exemples il faut citer : un nouveau traitement thermique pour l'alliage 7150, un nouvel alliage 7055, et de nouveaux traitements pour le contrôle de la composition et de la microstructure, par exemple le 2524. Des nouveaux alliages ont été développés, par exemple l'alliage aluminium-lithium à haute résistance 2195 ; l'alliage aluminium-lithium 2097 résistant à la fatigue ; des hybrides stratifiés d'alliages d'aluminium et de titane ; le Timetal 1100 pour des températures allant jusqu'à 1100 °F ; l'alliage Ti62222 ; des versions de Ti-6Al-4V à teneur en oxygène réduite ; de nouveaux alliages bêta titane ; des matériaux composites à matrice métallique améliorés ; des aciers anticorrosion à haute résistance ; des alliages Ni-Co améliorés et des aciers à bas carbone.

Ces nouveaux matériaux et traitements devraient permettre de prolonger de façon appréciable la durée de vie des aéronefs en service depuis longtemps qui composent l'essentiel des forces opérationnelles de l'OTAN. Le KC-135, dont l'entrée en service date de plus de 40 ans, le F-15, avion de supériorité aérienne, opérationnel depuis 20 à 25 ans ; et le F-16, et le KC-10 entré en service il y a au moins 15 ans. Bon nombre de ces appareils devront rester en service pendant encore 25 ans au moins. Leur maintenance est tributaire de considérations de sécurité et d'ordre économique. Le remplacement de composants endommagés par des matériaux avancés permettrait de diminuer les coûts globaux de possession de façon considérable. L'OTAN doit réfléchir à l'utilisation de ces nouveaux matériaux et l'atelier proposé représente un forum unique pour la discussion et la promotion continue d'approches communes du problème de la conservation des flottes aériennes internationales.

Publications of the RTO Applied Vehicle Technology Panel

MEETING PROCEEDINGS (MP)

Design for Low Cost Operation and Support
MP-37, Spring 2000

Structural Aspects of Flexible Aircraft Control
MP-36, Spring 2000

Aerodynamic Design and Optimization of Flight Vehicles in a Concurrent Multi-Disciplinary Environment
MP-35, Spring 2000

Gas Turbine Operation and Technology for Land, Sea and Air Propulsion and Power Systems (Unclassified)
MP-34, Spring 2000

New Metallic Materials for the Structure of Aging Aircraft
MP-25, April 2000

Small Rocket Motors and Gas Generators for Land, Sea and Air Launched Weapons Systems
MP-23, April 2000

Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft
MP-24, January 2000

Gas Turbine Engine Combustion, Emissions and Alternative Fuels
MP-14, June 1999

Fatigue in the Presence of Corrosion
MP-18, March 1999

Qualification of Life Extension Schemes for Engine Components
MP-17, March 1999

Fluid Dynamics Problems of Vehicles Operation Near or in the Air-Sea Interface
MP-15, February 1999

Design Principles and Methods for Aircraft Gas Turbine Engines
MP-8, February 1999

Airframe Inspection Reliability under Field/Depot Conditions
MP-10, November 1998

Intelligent Processing of High Performance Materials
MP-9, November 1998

Exploitation of Structural Loads/Health Data for Reduced Cycle Costs
MP-7, November 1998

Missile Aerodynamics
MP-5, November 1998

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Measurement Techniques for High Enthalpy and Plasma Flows
EN-8, April 2000

Development and Operation of UAVs for Military and Civil Applications
EN-9, April 2000

Planar Optical Measurements Methods for Gas Turbine Engine Life
EN-6, September 1999

High Order Methods for Computational Physics (published jointly with Springer-Verlag, Germany)
EN-5, March 1999

Fluid Dynamics Research on Supersonic Aircraft
EN-4, November 1998

Integrated Multidisciplinary Design of High Pressure Multistage Compressor Systems
EN-1, September 1998

TECHNICAL REPORTS

Recommended Practices for Monitoring Gas Turbine Engine Life Consumption
TR-28, April 2000

Verification and Validation Data for Computational Unsteady Aerodynamics
TR-26, Spring 2000

A Feasibility Study of Collaborative Multi-facility Windtunnel Testing for CFD Validation
TR-27, December 1999

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Summary

Due to the political situation the number of presentations was reduced from 15 to only 6, and the audience for each session was smaller than expected, only 25-30. However, a very high quality of papers was maintained and the technical content covered new alloys, processes, composites and coatings, including both military and civil experience. Additional time was available for discussion of each paper and the subsequent 'round table' discussion.

Advanced materials and processing methods are providing new options for materials selection and component design. The papers presented showed how new metallic alloys and composite materials can improve the performance and life of airframe structures. Larger units can be manufactured by improved processes such as investment casting, welding, creep-age forming and high-speed machining. This reduces the parts count and reduces assembly costs. Life can be extended by new corrosion resistant coatings, but these must be environmentally friendly.

Round Table Discussion.

The discussion raised the important issues of NDE, inspectability and damage tolerance. Larger structures may be easier to inspect since they have less fasteners, less holes etc. However, large unit constructions do not contain the crack stoppers inherent in smaller riveted components. This must be considered carefully in design. Also the load path during failure will be different in these new units so substitution needs careful stress analysis and full understanding of the failure modes.

However, these large parts will undoubtedly require coatings to protect them from corrosion and care should be taken to ensure that this does not impair inspectability. Discussion raised the point that civil aircraft experience more corrosion damage, although they are usually manufactured from similar materials. For example, Dassault Aviation see more corrosion problems in business jets than in the Mirage, but business jets fly about twice as many hours per year. British and Canadian experience has shown a high incidence of corrosion in transports and maritime patrol aircraft, although this is often linked to poor design and coating technology which reflects the age of the systems.

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A change to a system of 'Qualification by Analysis' would provide the step change necessary for materials substitution. This topic has already been raised as a potential AVT Technical Team activity under ageing systems. Certifying authorities insist on full structural tests for new flight-critical components. However there is already some shift away from testing since new versions of aircraft can be approved if they are extrapolations of existing designs and have appropriate laboratory and analytical data to support the new design. Numerical stress analysis of designs is now relatively easy and accurate. However there is still poor quantitative understanding of the effects of ageing and the definition of failure. Qualification by analysis will need a full identification of all the possible failure modes and a risk assessment which quantifies the most important areas of concern. These may change with time as the aircraft role is changed and if damage, such as corrosion or inelastic strains start to affect the load paths in the structure. These are issues which should be addresses in the life cycle management of the system.

AGING SYSTEMS IN AERONAUTICS AND SPACE DAMAGE TOLERANCE IN HELICOPTERS

Paolo Santini
A Keynote Lecture
Spring 1999 Meeting on Aging Systems
North Atlantic Treatise Organization
Applied Vehicle Technology Panel
Corfu, Greece, April 1999

1 INTRODUCTION

The aim of this keynote lecture should be that of introducing the main (or, at least,the common) features of the papers that will be presented during the Workshops that will take place in Corfou this week: Aging Aeronautical Systems, Life Extension for Helicopters,Propulsion and Gas Generators for aerospace vehicles.

A keynote lecture may be given in several ways. The most obvious form is probably an overview of the subjects that are to be discussed, trying to connect them together, so as to prepare the audience to a more detailed knowledge of them. However, in this way one runs the risk of saying something that will be repeated much better by the specific Authors, (and, often in a much poorer way than the Authors themselves), also because the time is in general very short.

Another way is to address and illustrate forthcoming needs and future trends in the area of interest: for this kind of presentation it would be necessary to be able to predict the future, because it happens very often that the future is not so smooth as everybody may imagine at the time of the lecture, and so, after a few years, or even a few months, needs and trends are completely different: a good example is the continuous series of war actions in the Balkans and in the Middle East, that may have a strong impact on the policy of the aeronautical production.

I have chosen another approach. When AGARD was established, I had the chance to take part in the first meeting in 1952, and I can well remember the great expectations resting on the new structure. I then followed AGARD throughout the decades, attending many meetings of the Structures and Materials Panel, until I became Panel Member in 1975 and, eventually, Panel Chairman during the period 1986-1988.

Paper presented at the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

I think therefore that it might be of some interest to look to the past, in order to trace back the historical development that led from the early scope of AGARD to the current situation.

2 THE HISTORICAL ROLE OF THE TECHNOLOGY

In order to understand clearly the development of AGARD we must before all else try to recall some general circumstances, summarizing them in two axioms.

The first axiom is that HISTORY IS DRIVEN BY TECHNOLOGY. This stems from two historical facts.First of all, political supremacy is retained by the States and by the political entities having a technological superiority with respect to other groups.

There are several examples of this in history. The Hittites of Asia Minor were a very small nation, but they were subjugating and overwhelming some much greater political groups, since they were able to master iron metallurgy, so that their carts were definitely superior to those of other surrounding populations. According to some historians, the military superiority of the Spartans was in part due to the fact that the waters of their rivers were colder than those of other regions of Greece, so that their quenching gave better manufacts. The Romans had a very efficient technology, in particular in civil engineering; they were first-class bridges and roads constructors, so their transportation systems was exceptionally well organized and allowed them rapid strategic movements of great armies. More recently Germany had a great advantage over the Allied Forces at the beginning of the war, and when the technological supremacy passed on the other side, Germany lost the war. And from this we learn a lesson: technology is not only a matter of scientific advance, but also of capability of mass production and of organization.

However the effect of technology on history is not confined to military superiority only. Its effect upon the quality of life is even more evident. Again here we have several examples. The so-called neolithic revolution, producing a much improved technology of stone artifacts, caused better methods for the stone-age man to acquire food, to cook it, to have better shelters, whose consequence was a lengthening of the average life expectation. Some historians consider the replacement of the double rigid yoke by the single flexible yoke in the Middle Age as a very import technological advance: as a matter of fact, it caused an increase of agricultural efficiency by a factor of 3 to 5.

There is no need to emphasize the scientific and technological discoveries of the Renaissance, or the tremendous progresse that created a completely new world in the 19.th century in the wake of the Industrial Revolution. And what about present times where we have a continuous increase not only of technological progress but of its rate of growth. We live today in a way which is completely different from what it was fifty, forty, and even thirty years ago: long trips are possible almost for everybody, every

family has one or two cars, TV and radio keep us informed in real time of the most extraordinary events: ladies at the age of 50-60 years are still charming (due to the much reduced work at home caused by modern appliances), and so on.

The second axiom is : TECHNOLOGY IS NOT CULTURE. By no means do I mean that the level of Technology is lower, but simply that Technology and Culture are two different expressions of the human imagination, and must be treated and considered differently. Culture is a free expression of what the mankind feels, either by reaction to the external world, or through an internal elaboration: the best examples for the latter being music (whatever music) and mathematics. Neither stem from external stimuli.

By contrast, technology cannot be free; it stems from human needs although in many cases industry creates those needs and imposes them: fifty years ago, nobody missed TV broadcasting, today nobody could live without it.

Technology is not applied science, but an application of science, which is culture. That is the difference and there is a continuous transformation moving from science to technology: a vivid example is provided by the history of spaceflight, synthetized by this sequel: Tsiolkowsky, Oberth, Goddard, von Braun.

3 THE DEVELOPMENT OF AGARD

AGARD seems to have been well aware of, and to have applied, the two axioms above.

In 1949, in connection with the 'warmest' period of the 'cold' war, NATO was founded (called at that time the Atlantic Pact) with a declared feature of military deterrent as a reaction to the increasing armament of the Eastern Block. The first axiom was immediately clear to the top men of the Western World; and, of course, one of the main concerns was relevant to Aeronautics (Space at that time was almost unknown). For this reason, some of the top scientists of the NATO countries were asked to meet and to propose which kind of organization would have been most adequate to improve the level of research and of technology in the aeronautical field in western country.

There was a large meeting in Rome at the end of 1952, attended by a great number of scientists; among them probably the most outstanding one was Theodore von Karman, a man of great scientific prestige and great cultural level. A typical mitteleuropean gentleman, he was the Chairman of the Meeting. I am proud to say: I was there, and I took part, although unofficially, in the foundation of AGARD. During those days, I had the opportunity to talk several times with Karman; especially at lunchtime we often sat together with Professor Broglio, my mentor, and Karman illustrated us his ideas on what the new organization should have been like. In industry at that time there was very lit-

tle research, also because the requirements were not so stringent as today: and he thought that a field such as aeronautics, in order to be competitive, should have been fed by science, by culture. In thinking so, he applied the second Axiom.

Theodor von Karman was a volcano: every time we met he came out with new ideas. I saw him personally drawing on a blackboard the word AGARDograph: he drew up a first draft of the future organization; he also chose the men who should have been the leaders of AGARD, and most of them were scientists or University Professors. Let me remember just some of them: Frank Malina, Frank Wattendorff, Luigi Broglio. Antonio Ferri. Karman transferred the same imagination to the other bodies he created in subsequent years and which are still in existence and florishing: ICAS (International Council for Aeronautical Sciences), IAF (International Astronautical Federation), IAA (International Academy for Astronautics).

The following year AGARD was formally established and the first Panels were formed. I remember the early steps of SMP, in 1955 with Broglio as Chairman, and four members only; among them was Thurston, recently dead. A great number of Members of the Panels came from the Universities, and brought their University mental attitude into the activities which were being proposed.

Thus the birth of AGARD was, in agreement with Karman's ideas, almost an academic institution, where basic research had a very important role. The 50's and the 60's saw a terrific development of the basic sciences of Aeronautics, and the beginning of the attention for Space. I will give just some examples. In 1957 there was a famous Meeting of SMP in Copenhagen where the foundations of modern structural dynamics were laid down. In 1960 a Meeting of the same SMP in Rome was one of the first Meetings dealing with the effect of kinetic heating upon structures, with special attention to reentry: and it should be remembered that we were in the very era of space challenge between USA and URSS, with the latter well ahead of the former. And, again, a basic step in Fluidodynamics was the famous meeting of FDP in Scheveningen, Netherlands, with the first results and methodologies in hypersonics, again with an eye to Space; of particular interest were the discussions and the forecast for future high speed experimental facilities. And let me also remember the wonderful results in the field of Aeroelasticity, an activity that lasted for decades, and that became a kind of permanent institution in SMP: and the great cooperative effort in this area, with the contribution of all the Nations under the leadership of giants such as Ashley, Kussner, Mazet; the discussions among them became a kind of social event!

I was appointed an official Member of SMP in 1975, and since then I took an active role in proposing activities: I must say that AGARD was for me an invaluable vehicle of contacting outstanding personalities in the aeronautical world. Also, my Department and my University took advantage,at a large extent, from the exchange of ideas and of personnel through

the Support Program which I believe was very beneficial not only to the supported nations but to the supporting nations as well.

However I noticed that the twenty years elapsed since the establishment of AGARD had basically changed the objectives and the general frame. Most of the Panel Members came from industry or from Air Force, and the subjects of investigations had gradually shifted towards applications: or, if you wish, towards Technology or Engineering. I consider that a very positive circumstance, since people like me were put directly in contact with the realities of the world of production. At every Meeting I learned a lesson; and I hope that some of my colleagues of the Panel also learned something from me.

I proposed several activities in SMP, also gradually shifted from science to Technology. Not everything, of course, was quite satisfactory: one of my criticisms concerned the fact that an Aero- Space Panel had very minor Space Activities. This was true also of other Panels. I was invited once to chair a session in a meeting of FMP centered on Space Flight and I was surprised to discover that the subjects were not as up to-date as I would have thought.

As a Panel Deputy Chairman I was for two years the Chairman of the Program Committee, and thus I had the opportunity to have a clearcut idea on what the general trend was,with an increasing shift towards more and more applied technology. F.i., I was rather surprised when a subject like 'Painting Removal' was proposed. Now I am well convinced that Painting Removal is of prime importance in everyday aviation practice. This is an example of what I believe was the invaluable contribution of AGARD.

In today's world much progress has been made in the past to improve our products. Now we also have another need: preserve what we have acquired, avoiding wast. AGARD and, subsequently, RTA have followed such requirement; this is the reason for the present Meeting on aging aircraft.

4 AGING AIRCRAFT

The age of the current aeronautical fleet can best be appreciated by referring to statistical value. Most of them refer to US data, which are probably the most updated and complete.

In 1997, 46% of the commercial aircraft were over 17 years of age, 28% over 20 years: the average age for the major companies is reported in Tab.I (from Ref.1).

TAB:I

Airline	Average Aircraft Age (years)
Alaska	7.6
American	10.0
American West	11.0
Continental	14.4
Delta	12.2
Northwest	19.9
Southwest	8.8
TWA	17.0

United	10.8
US Airways	12.8

The expectations for the near future are of an increase of 70% in the next ten years, with a total of 12,000 new airliners to meet the demand in 2015, for a typical design life of 60,000 cycles. Probably already in 2001 2,500 US commercial airliners will be flying beyond the original life span.

The definition of 'aged' or 'aging' structure and/or system would need a clear definition, which, on the contrary, is rather vague.We report, (from Ref.2) two definitions as provided by ESA: Aged Structure - A structure which may have structural degradation or damage as the result of being exposed to the combined effects of the environment.

Aging - The process of the effect t on materials of exposure to an environment (elevated temperature, ultraviolet radiation, moisture or other hostile environment) for a period of time. The problem of aging aircraft has attracted the attention of civil and military authorities since more than a decade. In general, the attention and the need for new studies are prompted by some spectacular accidents, which have a strong impact on the public opinion.

In 1988 a Boeing 737 of the Aloha Airlines, 19 ys.old, suffered from an accident caused by a fatigue failure in a panel of the fuselage: a flight attendant was killed, and 171 passengers were injured .This accident was almost immediately followed by an action from FAA who established the NAAR (National Aging Aircraft Research).

In 1996 another most famous accident occurred with the flight 800 TWA: the airplane was a Boing 747, 25 years old, 90000 flight hours, 18000 cycles, vs.the original design life of 60,000 hours, and 20,000 cycles (Ref.(1)). In this case the cause was attributed to an electrical malfunction followed by a spark in the fuel compartment. This was followed by a five year project to investigate and check nonstructural parts of the airplanes, such as wiring, hydraulics, avionics, etc. Especially wiring seems to need careful maintenance and inspections (almost zero until few years ago). Now in a commercial airplane there are miles and miles of wires and careful and systematic inspections may cost several million dollars to a Company. Yet they are necessary.

On the other hand, it must be recognized that maintaining old aircraft in service is a definitely positive factor from an economic viewpoint; f.i., the current cost of an Airbus 300 would be of the order of 100 million dollars, while refurbishing an old airplane of a comparable capacity could only cost 4 to 5 million dollars.

Very similar problems apply to military fleets also. F.i., the United States Air Force has many old (20 to 35 years) aircraft that are the backbone of the total operational force, some of which will be retired and replaced with new aircraft. However, for the most part, replacements are a number of years away: for many aircraft, no replacements are planned and many are expected to remain in service for another 25 years or more. Such aircraft have encountered or are considered likely to encounter aging problems,

such as fatigue cracking, stress corrosion cracking, corrosion and wear.

For these reasons, Committees and Working Groups were formed within the Air Force scientific framework, with the following objectives: (i) Identify and correct structural deterioration that could threaten aircraft safety; (ii) prevent and minimize structural deterioration that could become an excessive economic burden or could adversely affect force readiness, in terms of performances; (iii) predict. for the purpose of future force planning, when the maintenance burden will become so high or the aircraft availability so poor that it will no longer be viable to retain the aircraft in the inventory.

The Committee arrived at Near Term and Long - Term Research Recommendations, and produced a rather lengthy list of items to address the attention to. This list can be found in Ref.(3) (it would be too long to reproduce it here): looking at it, we can see that there is room not only for technology, but also for basic research, and for items that are prone to be completely re-invented.

It is very instructive to have a look at the items of the list. First of all we have the confirmation of the prime role of corrosion in defining the lifetime of an airplane.This well known to everybody, except perhaps for the 'size' of this role. As a matter of fact, corrosion is a social problem : it costs about 100 billions USD per year to United States, almost 300 dollars per citizen. And USAF is responsible for about 10% of this figure. Furthermore we see other fields to which, in my opinion, insufficient attention has hitherto been given by fatigue people, such as structural dynamics and dynamic response. But it must be a completely new structural dynamics, not based on the conventional analitycal methods, eigenfrequencies and modes, but rather on the need of predicting stress concentrations arising from dynamic effects and following cracks under dynamic conditions. I am sorry for not being able to illustrate other important suggestions that can be derived from the list for lack of space.

But the subject of aging aircraft is so important that there is a sure need in the future of specialized technical staff in this area. Appropriate courses of formation have been designed, with the aim of providing the attendees with the necessary knowledge: an example is given in Ref.(4-5). Specialists in aging airplanes will become in the future a new professional profile.

5 DAMAGE TOLERANCE ISSUES

The life of a structural component is limited by the effects of its usage history, which may consist of cyclic loads, fluctuations in temperature,etc. In the past, and also today , several design philosophies have been introduced into current practice, and among them damage tolerance has probably become one of the most prominent in aircraft industry (Ref.(6)).

I shall not dare to give a general overview on what damage tolerance concept is. I just will confine myself to recall - or try to imagine- why

there is a special need to treat damage tolerance as applied to helicopters in a very special way. As said, the life of a structural component depends on its usage, which varies from one type of flying machine to another. There is however a general agreement that critically loaded components such as engine rotors or landing gears can have shorter replacement intervals. In any case, the usage profile must describe the various flight conditions, and the amount of time spent at each gross weight, speed, and altitude, which define the load factor spectrum, and subsequently dictate the inspection intervals.

There is no need to illustrate how much the environmental spectra for an helicopter is different from that of an airplane. Considerable effort is needed on loads generation, and determination of material data. Also, this is a field where theoretical investigation must also be accompanied by experimental data.

Let me clarify this concept in some detail. In the past there has always been a dichotomy: theory vs. experiments, the former being looked at rather suspiciously. The two activities were almost independent of each other, and the optimum was reached when there was a satisfactory (not always well-defined) agreement between the two results. With the advent of computers, 'theory' has been replaced by 'mathematical model'and everybody is happy, but the dicotomy has not yet been fully overcome. It is my opinion that the two activities must be viewed not from the point of 'mutual check' but from that of 'mutual integration'. Experimental data for a specific problem or design must supplement mathematical models, providing information that no theory whatever can provide : theory must provide detailed information that no experiment can provide.Although the last statement is more difficult to accept, there are several examples validating it (e.g., in Fluid Dynamics).

And again, I must say, damage tolerance in helicopters is a good example on how increasing industrial needs stimulate basic research.

6 PROPULSION

I am not an expert in Propulsion: I am simply an admirer of it. I have already mentioned the names of the pioneers of Spaceflight, which is one of the top conquests of man : for all of them Spaceflight was identical with Propulsion. And, as a matter of fact, space activities only became possible when sufficient thrust was made available to scientists. This is very often, not to say too often, forgotten today.

Most recently, because of economic problems, as happens more and more frequently today, the subject of small rocket motors has become increasingly popular. In this Symposium attention is focused on the specific goals of NATO and on the relevant applications. All the main features of the problems are examined with respect to weapon systems. And here a further remark is appropriate: in the history of technological development, military applications have very often led the way to a more general use of the progress achieved.My hope and my message is that this will be the case also for this

Conference.

7 CONCLUSION

We have two main conclusions to draw. The first that we are at an era in which we rediscover a new feature of Technology: the Technology of restoring, refurbishing, reusing the system created by the human imagination in the recent past. Second, that such operations of re-whatever we must use Science and Culture again, although in a different (and probably more advanced) way than in the past.

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THE NEED FOR NEW MATERIALS IN AGING AIRCRAFT STRUCTURES

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SUMMARY

The end of the Cold War and political and economic considerations has resulted in an effort to extend the life of many aircraft that are the backbone of NATO operational forces. Although some are designated to be replaced with new aircraft, the replacement schedule of many often requires an unprecedented life span of between 40 to 60 years before retirement. Many of the older aircraft have encountered, or can be expected to encounter, aging problems such as fatigue cracking, stress corrosion cracking, corrosion and wear. In order to ensure continued airworthiness and flight safety the structural components undergoing these problems will have to be repaired or replaced. Alloy development that has taken place since a large percentage of the older aircraft were put into service has resulted in several new materials, heat treatments and processing technology that can be used for appropriate longer lasting and higher performing airframe components thus reducing life-cycle costs. This paper describes some of these materials and their advantages over those suffering from "aging problems."

1. INTRODUCTION

In 1996 the U.S. Air Force requested the National Research Council (NRC) to identify research and development needs and opportunities to support the continued operations of their aging aircraft. The results of this study, which was undertaken by a committee selected by the National Materials Advisory Board of the NRC, were published in the Committee's final report in 1997.[1] Among the many recommendations made by the Committee, one was to develop guidelines to broaden the application of improved materials as substitutes for incumbents with low damage tolerance and corrosion resistance. Such substitutions must make good business sense with respect to reduction in life-cycle costs and materials availability. Examples of reducing the life-cycle cost by implementing new materials on aging aircraft structure are given in the paper by Austin et al in this proceedings.[2]

The U.S. Air Force, as well as the air forces of other NATO countries, has many old aircraft that form the backbone of the total operational force structure. Many of these, e.g. the KC-135, the B-52, and the C-141were introduced into service in the 1950s and 1960s. Even the F-15 air superiority fighter became operational 20 to 25 years ago and the F-16 and KC-10 jet trainer at least 15 years ago. The extended use of these aircraft,

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in conjunction with frequent changes in mission requirements, results in increased maintenance and repair costs associated with structural cracking and corrosion problems which are, in most cases, associated with aluminum alloys and tempers developed prior to 1960.

Structural (fatigue) cracking is a direct result of aircraft use; i.e. load or stress cycles, and will eventually occur in all aircraft. Corrosion results from the exposure of susceptible materials to various corrosive environments, e.g. humid air, saltwater, sump tank water and latrine leakage, and to inadequate or deterioration of corrosion protection systems. In the case of aluminum primary structure, numerous service difficulties have been documented on components manufactured from alloys 2024-T3, 7075-T6, 7178-T6 and 7079-T6. For example, in order to minimize structural weight and thus maximize payload capability of the KC-135, the Air Force elected to use 7178-T6 in the lower wing skins and 7075-T6 in other locations in the aircraft. These alloys were designed to emphasize strength and have low damage tolerance. In 1977 the Air Force recommended that the wings be redesigned using more damage tolerant 2024-T3 and also recommended cold working fastener holes in the remaining 7178-T6. However, there is currently concern about the long-term effectiveness of the cold worked fastener holes and structural deterioration of the 2024-T3 due to exfoliation corrosion and multi-site corrosion-fatigue damage.[1]

Research since 1960 has led to the development of several new aluminum alloys, heat treatments and processing methods that offer more damage tolerant and corrosion resistant alternatives for airframe components than those that were used in the older aircraft. The overaged T73 and T76 tempers were developed in the early 1960s to make 7075 more corrosion resistant to stress corrosion cracking and to exfoliation corrosion; however, the improvement obtained is at the expense of strength. In 1974 Cina obtained a patent[3] specifically targeted at 7075, for a heat treatment procedure to provide stress corrosion resistance equivalent to an overaged T73 temper while maintaining the peak-aged strength. Although the concept, called retrogression and re-aging (RRA), seemed industrially impractical at the time, derivative tempers have been taken to practice as will be discussed in the paper by Holt et al in this proceedings.[4] In the 1970s alloy 7050-T74 was developed to fill the need for a material that would develop high strength in thick section products, good resistance to exfoliation corrosion and stress corrosion cracking, and adequate fracture toughness and fatigue characteristics. Also, in the 1970s a derivative of 7075, i.e. 7475, was developed that provided improved fracture toughness compared with 7075. In the 1980s a new generation of low density Al-Li alloys, e.g., 2090, 8090 and 2091, was developed that offered alternatives, other than increasing strength, for reducing structural weight. During the past decade new improvements have evolved to address the alloy limitations found in pre-1980s aircraft and if used in retrofitting will result in maintenance schedules similar to that required for new aircraft.

The purpose of this paper is to review some recent advances in derivative alloys that have occurred primarily through the use of a very large scientific knowledge base and tighter chemistry and process controls. The newer alloys offer useful improvements in product

performance, quality and reliability and can be applied to aging aircraft problems to dramatically reduce maintenance costs.

2. RECENT ADVANCES IN DERIVATIVE ALLOYS AND TEMPERS

2.1 Improvements in Strength, Corrosion Resistance and Toughness

During the retrofitting of aging aircraft the substitution of alloys with equivalent strength but with higher corrosion resistance and fracture toughness will extend maintenance schedules, decrease down time, and reduce costs. As mentioned previously, RRA offered promise to achieve this goal. In the 1980s work by Wallace and co-workers[5-7] showed that beneficial retrogression and re-aging (RRA) effects can be obtained in large components if the retrogression temperatures are below 200° C for 7075. Hepples et al[8] showed that RRA 7150 using commercially viable thermal process routes can provide material with peak strength and high resistance to SCC and exfoliation corrosion. Based on the RRA concept, Alcoa developed the T77 temper for 7XXX alloys, e.g. 7150. The improvement in the increase in combination of strength/corrosion resistance via the T77 temper process is illustrated in Figure 1.

Alloy 7150-T77 has higher strength with durability and damage tolerance characteristics matching or exceeding those of 7050-T76. Boeing selected extrusions of 7150-T77 as fuselage stringers for the upper and lower lobes of the 777 jetliner because of the superior combination of strength, corrosion and SCC characteristics and fracture toughness. Alloy 7150-T77 plate and extrusions are also being used on the new C17 cargo transport.[9] Improved fracture toughness of 7150-T77 products is attributed to the controlled volume fraction of coarse intermetallic particles and unrecrystallized grain structure, while the combination of strength and corrosion characteristics is attributed to the size and spatial distribution and the copper content of the strengthening precipitates.[9] The improvement in properties using the new temper, relative to older alloys and tempers, is illustrated in Figure 2.

Alloy 7055 was developed by Alcoa for compressively loaded structures.[10] Alloy 7055-T77 plate and extrusions offer a strength increase of about 10% relative to that of 7150-T6 (almost 30% higher than that of 7075-T76). These products provide a high resistance to exfoliation corrosion similar to that of 7075-T76 with fracture toughness and resistance to the growth of fatigue cracks similar to that of 7150-T6. In contrast to the usual loss in toughness of 7XXX products at low temperatures, fracture toughness of 7055-T77 at –65°F (22K) is similar to that at room temperature. Resistance to SCC is intermediate to those of 7075-T6 and 7150-T77 products. A comparison of properties of these 7XXX alloys is given in Table 1. The attractive combination of properties of 7055-T77 is attributed to its high ratio of Zn:Mg and Cu:Mg. When aged to T77 this composition provides a microstructure at and near grain boundaries that is resistant to intergranular fracture and to intergranular corrosion. The matrix microstructure is resistive to strain localization while producing a high strength. Alloy 7055-T7751 is used for the skin of the upper wing surface of the Boeing 777. The improved strength-toughness properties of newer alloys and tempers, relative to the older ones, are illustrated in Figure 3.

2.2 Improvements in Damage Tolerance and Multiple Site Damage

The service life of an airframe can potentially introduce multi-site damage (MSD) states such as widespread fatigue or widespread corrosion that may imperil the structural integrity of the aircraft. For this case, the inspection intervals set by standard residual strength and damage tolerant design that are normally directed at the presence of a single crack, are inadequate. This realization and the desire for reliable. longer lasting aircraft with lower maintenance costs has given rise to requirements that non-pristine or aging structure be accounted for in maintenance strategies.[11] This philosophical shift creates the opportunity for affordable, replacement materials that can not only resist the occurrence of multi-site damage. but which offer improved structural damage tolerance when MSD is present.

The occurrence of widespread damage sites can be associated with the intrinsic characteristics of the material microstructure.[12] Material microstructural sites prone to the development of crack-like damage, attributable to corrosion or fatigue, can be associated with particles, inclusions, pores, and grain boundaries.[13] While these features are necessarily a part of the material, the character of these features can be altered through composition and process modifications while still meeting the required material strength performance characteristics.[14]

Machined structures from plate thicker than three inches is often used to reduce part count and assembly costs associated with built-up components manufactured from thinner material. However, since the thicker plate undergoes less work than thin products there is a higher probability that porosity developed during the casting operation will not be sealed. Obviously, the high porosity material has a poorer fatigue performance than low porosity material. There has been continuous process refinement in the production of thick plate since the early 1980s that has reduced porosity as well as particle and inclusion size. Consequently, the fatigue lifetime of products produced from the more recent material, even for a one-to-one substitution, should be longer than products produced from pre-1980 material. The effect of the process refinement on the fatigue lifetime of 7050-T7451 is illustrated in Figure 4.

Alloy 2024-T3 sheet is often selected for wing and fuselage skins for its superior damage tolerance properties when compared to higher strength 7XXX products. A derivative of 2024-T3, 2524-T3, was recently developed by Alcoa[15] and offers improvement in strength/fracture toughness (approximately 15-20%) and fatigue crack growth resistance (2X) over 2024-T3, Table 2. The improvement was achieved through very tight controls on composition and processing based on the knowledge that constituents associated with Fe and Si impurities lower fracture toughness[16-19] and have an adverse effect on both fatigue crack initiation[13] and fatigue crack growth resistance.[20] Coarse primary phases formed when solubility limits are exceeded at the solution heat treatment temperature (or those formed during hot rolling and not re-dissolved during subsequent processing) have a similar effect.[21] Consequently, tight controls on chemistry, i.e. low levels of Fe and Si, balancing the Cu and Mg content to produce maximum strength without exceeding solubility limits at the solution heat treatment temperature,[22] and a controlled processing schedule are all necessary.[21] In controlling the Cu and Mg contents, the levels of Fe, Si

and Mn in the alloy have to be considered since the constituent phases in 2X24 are usually Al_7Cu_2Fe, $Al_{12}(Fe,Mn)_3Si$, $Al_6(Fe,Cu)$ and the dispersoid is $Al_{20}Cu_2Mn$. The effect of fewer and smaller constituent particles on fatigue initiating corrosion pits is illustrated in Figure 5.

The fatigue crack growth advantage that 2524 has over 2024 enables an increase in operating stress, which offers a weight saving opportunity that may also accommodate mission changes that have occurred in older aircraft. This improvement also allows for an increase in inspection interval, which translates to lower operating costs. Inspections are easier since larger crack sizes can be tolerated and longer critical crack lengths translate to an increase in safety. The effect of skin alloy and operating stress on inspectable crack growth life is illustrated in Figure 6 for a longitudinal fuselage skin crack under an intact frame. Also, 2524 body skin offers substantial residual strength and cyclic life improvements over 2024 in multi-site damage scenarios, Figure 7. The fatigue advantage of 2524 over that of 2024 caries over to corroded material as illustrated in Figure 8. The higher toughness and greater resistance to fatigue crack growth of 2524 resulted in the elimination of tear straps in a weight-efficient manner on the Boeing 777.

2.3 Reductions in Density and Improved Fatigue Crack Growth Resistance

The second generation of Al-Li alloys (the first being the Alcoa alloy 2020) were developed in the 1970s (alloy 1420 in Russia) and the 1980s (alloys 2090, 2091, and 8090). The Al-Mg-Li alloy 1420 and the Al-Li-Cu-X alloys 2090 and 8090 are now in service in the MIG 29, the EH1 helicopter and the C17 transport. Alloy 1420 has only moderate strength and the Al-Li-Cu alloys (which contain approximately 2% lithium) have a number of technical problems, which include excessive anisotropy of mechanical properties, crack deviations, a low stress-corrosion threshold and less than desirable ductility and fracture toughness. Newer Al-Li alloys have been developed that have lower lithium concentrations than 8090, 2090 and 2091. These alloys do not appear to suffer from the same technical problems. The first of the newer generation was Weldalite 049® (2094) which can attain a yield strength as high as 700 MPa and an associated elongation of 10%. A refinement of the original alloy, 2195 which has a lower copper content, is now being used for the U.S. Space Shuttle Super-Light-Weight Tank. Alloy 2195 replaced 2219 and, along with a new structural design, saved 7,500 pounds on the 60,000-pound tank. This allows an increased payload for the Shuttle and reduces the number of flights necessary for the construction of the International Space Station, thus saving millions of dollars.

Three other recent derivatives of the third generation of Al-Li alloys are 2096, 2097 and 2197. They contain lower copper and slightly higher lithium content compared to 2024. Alloys 2097 and 2197 contain a very low Mg content to improve SCC resistance and Mn to prevent strain localization normally associated with the shearable Al_3Li present in the higher Li-containing alloys. Alloy 2097/2197 was recently selected[2] for replacing 2124, which had fatigue problems, for bulkheads on the F16. Alloy 2097 has a 5% density advantage over 2124 and at least 3 times better spectrum fatigue behavior or approximately 15% higher spectrum fatigue stress allowable. Although Al-Li alloys are more expensive than conventional aluminum alloys, the replacement of 2124 by 2097 for

the BL 19 Longeron of the F16 doubles the service life of the part, saving over twenty-one million dollars for the fleet of 850 USAF aircraft.[2] Engine access cover stiffeners, currently made from 2124, are also being replaced by Al-Li alloys due to their better fatigue life. This is an excellent example of retrofitting with improved materials for reducing life-cycle costs as described by Austin et al.[2]

3. SUMMARY AND CONCLUSIONS

Older aircraft can be retrofitted with new materials providing improved DADT when compared to the materials used during the original manufacture of older aircraft. A few scenarios for exploiting the potential benefits of new material replacements are given in Table 3. Continuous improved and derivative variants of existing alloys have the broadest utilization potential. Many of these materials are already flying on new aircraft, e.g., the Boeing 777 and/or have been used for retrofitting aircraft e.g. the F-16. Some alloys may be considered as "preferred equivalents" to their predecessors regardless of application, e.g., 2524 for 2024, and others may be considered "preferred replacements" within limits, e.g. 7XXX-T7X for 7075-T6. However, in order to facilitate retrofitting of aging aircraft with new materials, a generic material substitution system is needed for rapid/broad implementation of the *best* material solutions. This system should include ways to improve the efficiency of the substitution process by substantiating new materials as "preferred replacements," by approving the alloy substitution matrix, and by defining opportunities and cost/benefit trades for replacement scenarios. In addition, the repair and maintenance centers should stock qualified substitutes in order to reduce down time for retrofitting.

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Table 1. Longitudinal Property Comparison in One Inch 7XXX Plate

Alloy Temper	UTS (ksi)	TYS (ksi)	CYS (ksi)	El (%)	Kic ksi(in)$^{1/2}$	ASTM Exco rating	SCC ASTM G47@ 20 days (ksi)
7075-T651	76	69	66	6	20*	ED*	10*
7150-T7751	84	78	77	8	22	EB	25
7055-T7751	89	86	85	7	21	EC	15*

*** Typical**

Table 2. Typical Mechanical Properties for 2524-T3 and 2024-T3 Sheet in the Long-transverse Direction.

Alloy	Thickness (mm)	UTS (MPa)	TYS (MPa)	Elong (%)	Kc[a] MPa-m$^{1/2}$	da/dN@ Δ K=33[b] (mm/cycle)
2524-T3	0.81 - 1.59 1.60 - 3.26 3.27 - 6.32	420 441 441	303 310 303	19 21 22	174	2×10^{-3}
2024-T3	0.81 - 1.59 1.60 - 3.26 3.27 - 6.32	427 448 448	296 310 310	18 19 19	141	6.9×10^{-3}

a) M(T) specimen, T-L orientation, W = 40.6 cm (16 inch), $2a_0$ = 10.2 cm (4 inch) tested per ASTM B 646.
b) T-L orientation, tested per ASTM E 647 under constant Δ K conditions, R = 0.1, relative humidity > 90%.

Table 3. Possible Scenarios for Exploiting the Potential Benefits of New Materials

Repair Option	Primary Requirement	Potential Benefits	Potential Disadvantages	Time/Risks Resources
Identical component/ material replacement	Maintain safety & get it flying	Straightforward	"Prolongs the agony" with high repeat repair costs	Lowest
Form-fit function (material upgrade)	Reduce cost of maintenance, improve readiness	Some capture of new materials benefits	Requires M&P, design and analysis expertise	Moderate
Re-optimize with material upgrade	All of the above plus performance	Greater capture of new materials benefits	Requires extensive M&P, design, fabrication, analysis expertise	Moderate to high
Total redesign with new concept	Maximize life-cycle economics & performance	Full capture of best available technology	Requires full OEM capabilities	Highest

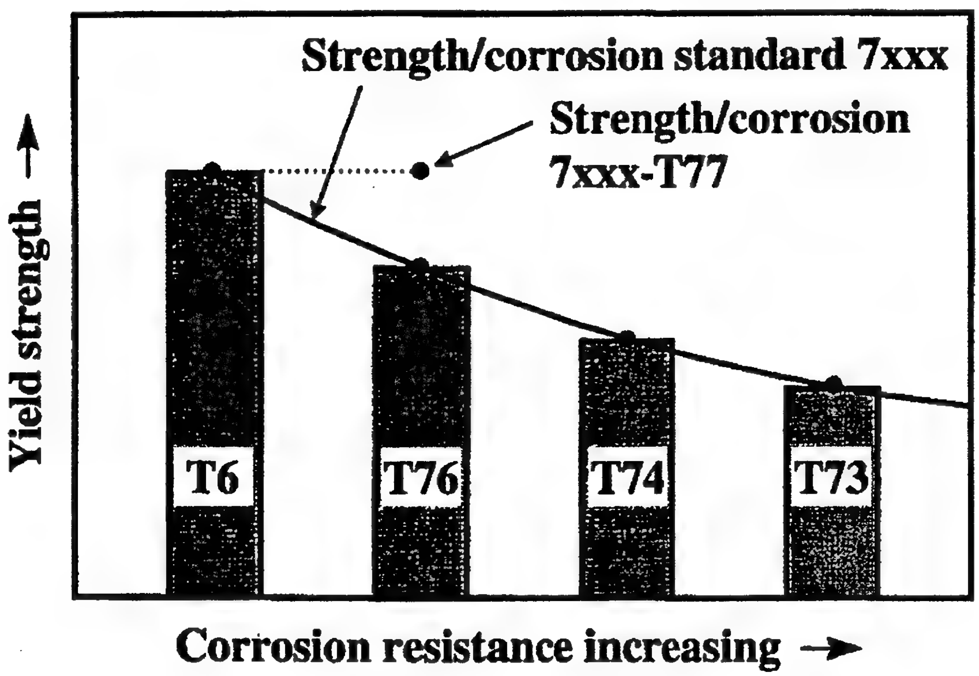


Figure 1. Improvement in strength/corrosion combination due to the T77 temper.

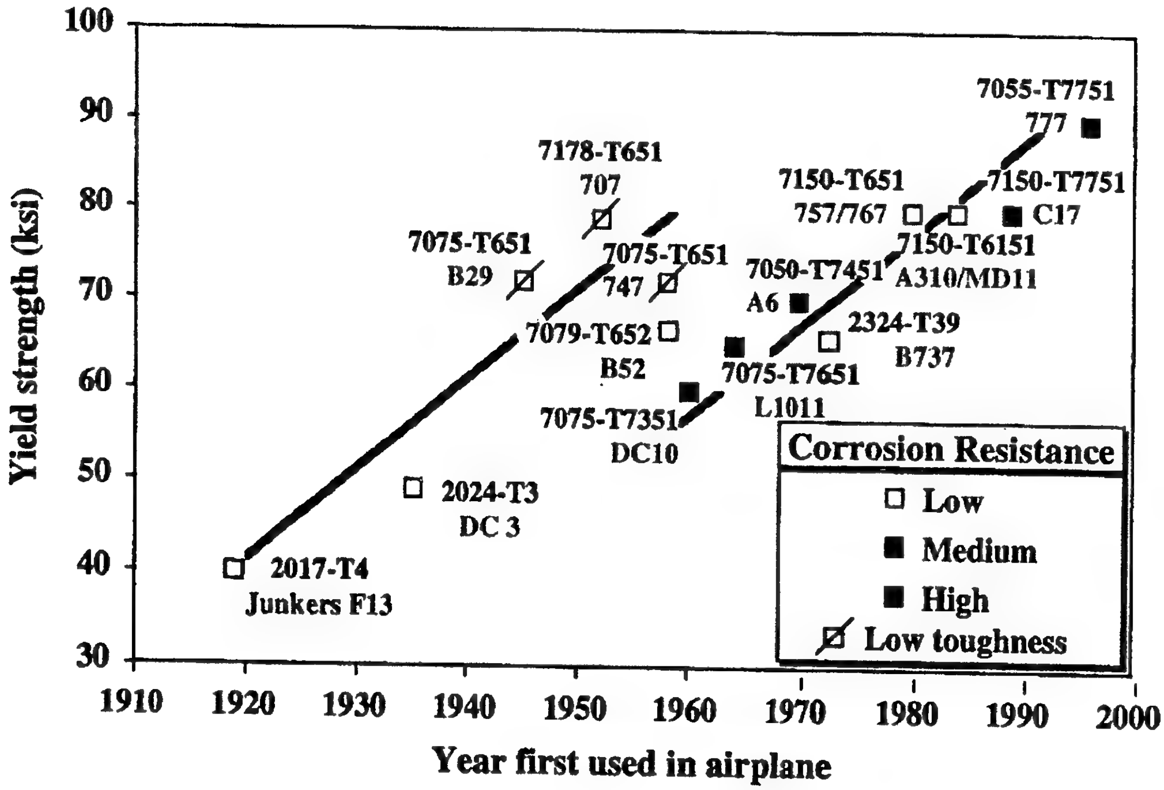


Figure 2. Comparison of strength and corrosion resistance of various aluminum aerospace all as a function of year first used in airplane.

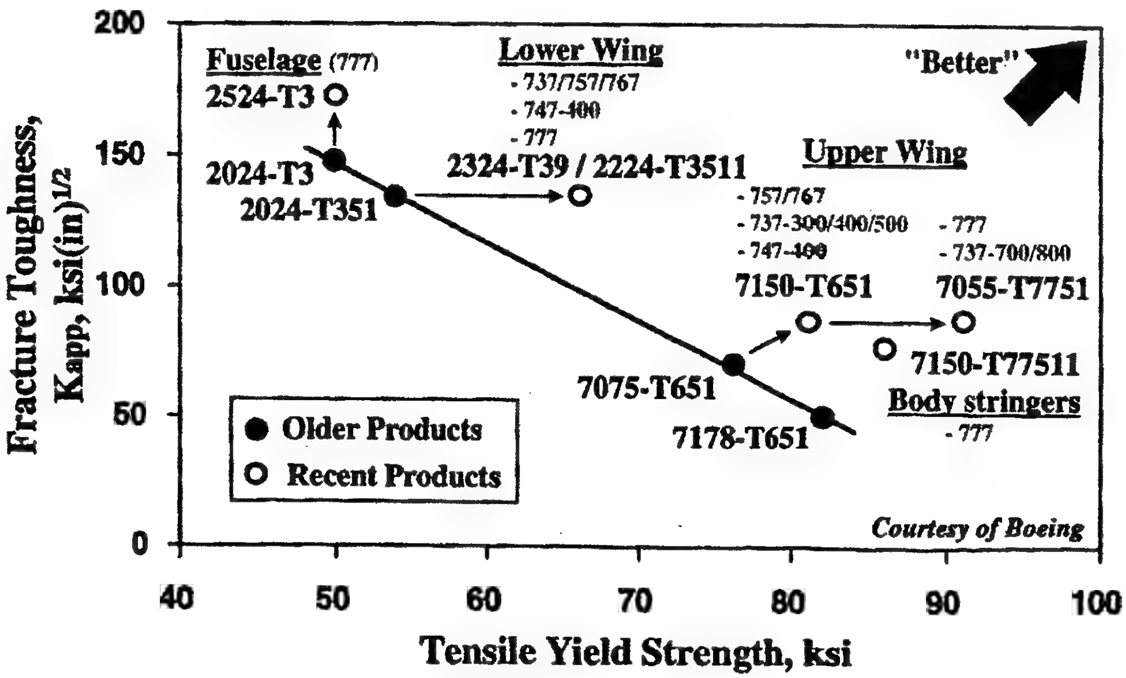


Figure 3. Comparison of fracture toughness/yield strength of older products with newer aluminum products.

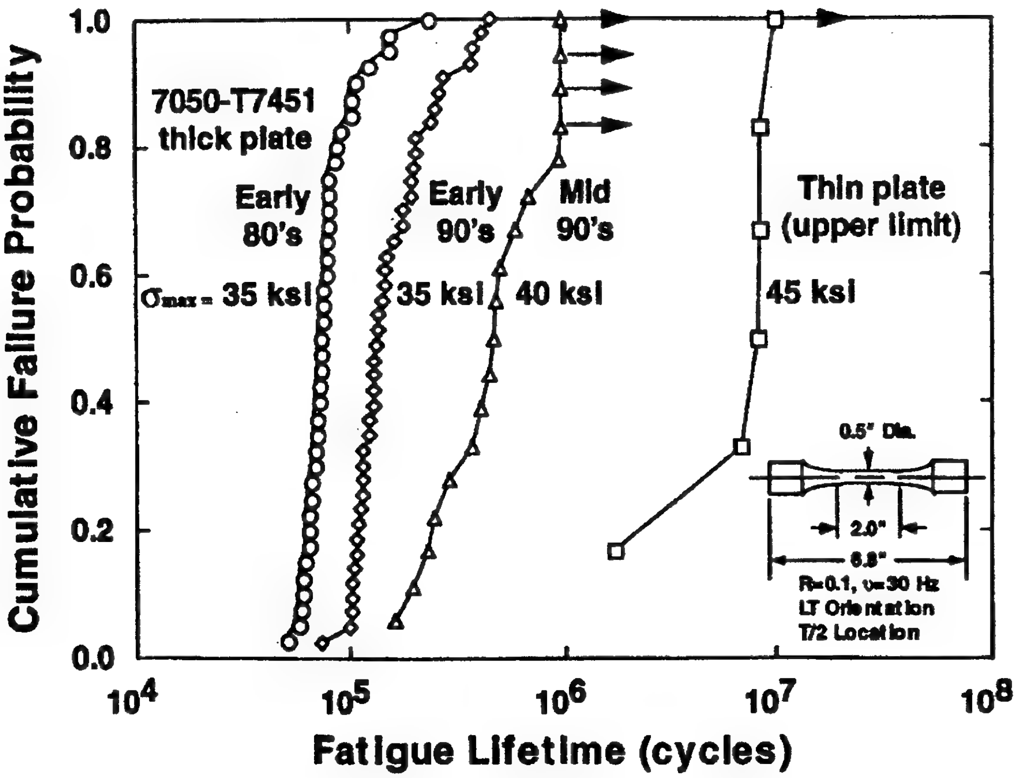
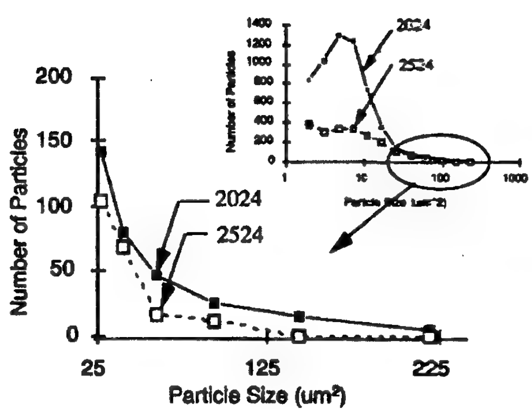
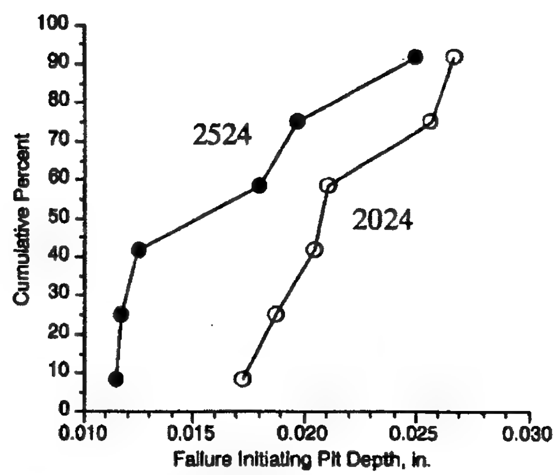


Figure 4. Improvement in fatigue lifetime of 7050-T7451 due to process refinement.



Comparison of constituent particle distribution in 2024 and 2524 sheet (0.125 in. thk., L-ST plane)



Cumulative failure-initiating pit-depth* distributions in 2024 and 2524 sheet
*** measured in plane of specimen fracture**

Figure 5. Effect of constituent particle distribution on fatigue-initiating pits.

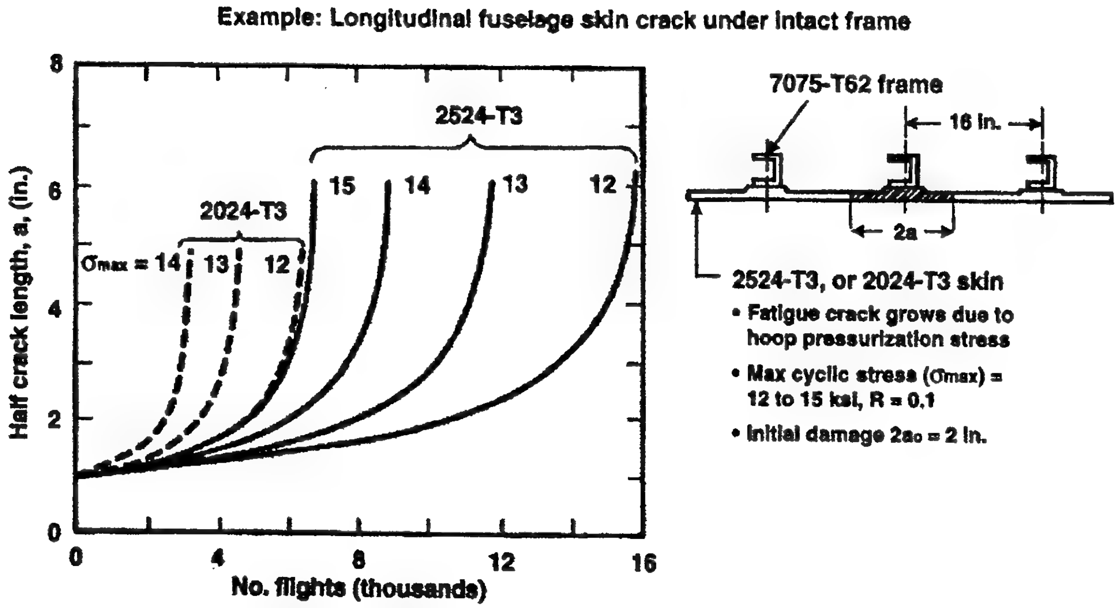


Figure 6. Effect of skin alloy and operating stress on crack growth life.

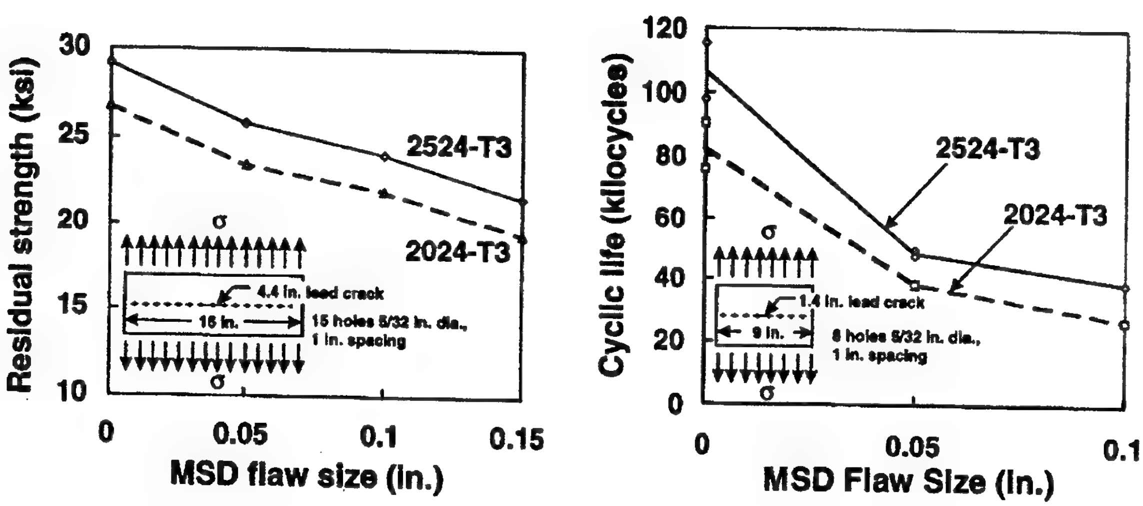


Figure 7. Residual strength and cyclic life capabilities of 2524-T3 and 2024-T3 skin sheet (clad, 0.05 in. thk.) in wide, multi-holed panels with central lead crack and varying size MSD cracks (two per hole).

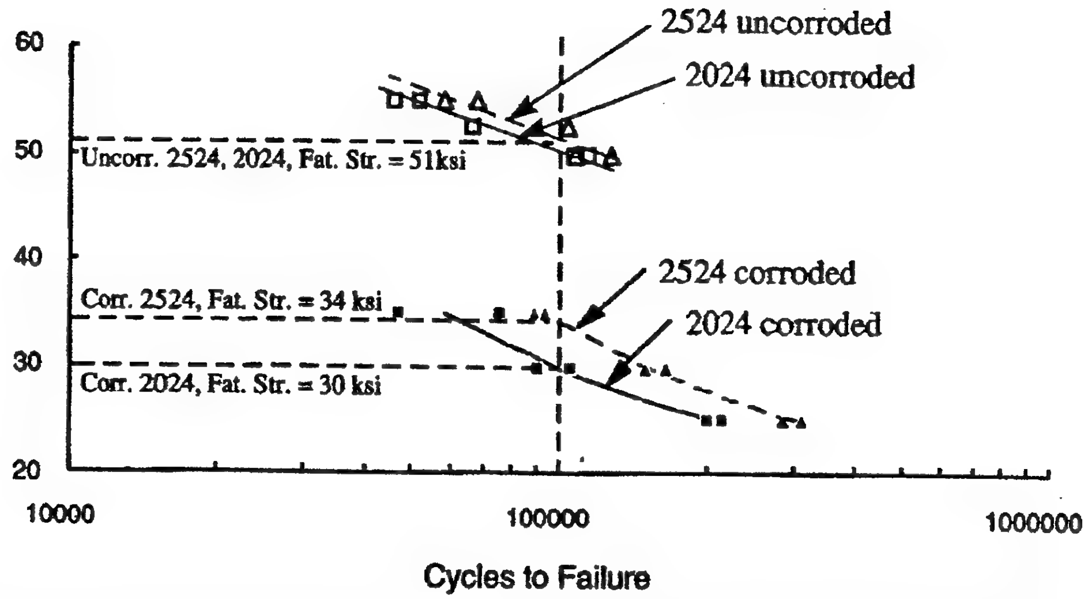


Figure 8. Axial S/N fatigue performance of 2024-T3 and 2524-T3 bare sheet (0.124 in. thk.) with and without prior corrosion.

Implementation of New Materials on Aging Aircraft Structure

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Numerous advances in new materials such as aluminum-lithium alloys, discontinuously reinforced aluminum composites, elevated temperature alloys, and other materials have occurred over the last several decades by academia, industry, and government laboratories around the world. However, implementation of these materials for primary aircraft structure have been infrequent due to several key issues, including alloy suitability for the required service environment, deficiencies in microstructure/properties, implementation timing, as well as end-user/customer acceptance of new materials. Two material systems that have been implemented through a team of Lockheed Martin Tactical Aircraft Systems, DWA Aluminum Composites,McCook Metals Ltd (formerly Reynolds Metals Co.), and U.S. Air Force engineers include 6092/SiC/17.5p-T6 Discontinuously Reinforced Aluminum (DRA) sheet and 2297-T861 Aluminum-Lithium plate. This paper provides a background on successful technology transitions in the commercial sector, descriptions of the development and building-block testing of the DRA and AlLi materials, and lessons learned on the successful implementation of these two materials on existing aircraft structure.

Introduction

High performance aircraft require a myriad of materials technologies to meet the performance, weight, and affordability standards required by the end-user. Over the past thirty years, the number of new materials systems that the materials industry designs has surpassed the available resources to qualify and/or implement these materials into aircraft. Some systems did not live up to the projected expectations. Early disappointments with aluminum alloys containing relatively high lithium contents (>2%) created a significant impediment to qualifying new lithium containing alloys. Similarly, reinforced powder metallurgy alloys experienced several early set-backs arising from scale-up and property shortfalls in the area of fracture toughness. High temperature aluminum alloys have not yet enjoyed the success of high volume usage on aircraft systems. In general terms, the road to implementation seems rather long, as illustrated in Table 1. ("Bringing New Materials to Market", Tech. Review, Feb/Mar 1995, pp43-49)

Paper prepared for the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

Table 0-1. 20 Years from Invention to Commercialization

Materials Technology	Invention	Widespread Commercialization
Vulcanized Rubber	1839	Late 1850's
Low-Cost Aluminum	1886	Early 1900's
Teflon	1938	Early 1960's
Titanium (Structural Uses)	Mid 1940's	Early 1960's
Velcro	Early 1950's	Early 1970's
Polycarbonate (Bullet Proof Glass)	1953	About 1970
Gallium Arsenide	Mid 1960's	Mid 1980's
Diamond-Like Thin Films	Early 1990's	Early 1990's
Amorphous Soft Magnetic Materials	Early 1990's	Early 1990's

With the materials in Table 1, it took on average twenty years from invention to widespread commercialization. Today, materials producers can expect a ten year cycle to fully qualify a material system for structural applications. As mentioned previously, several new systems that showed promise were dropped from implementation consideration. Figure 1 shows a schematic criteria map for materials from initial research and development through production implementation.

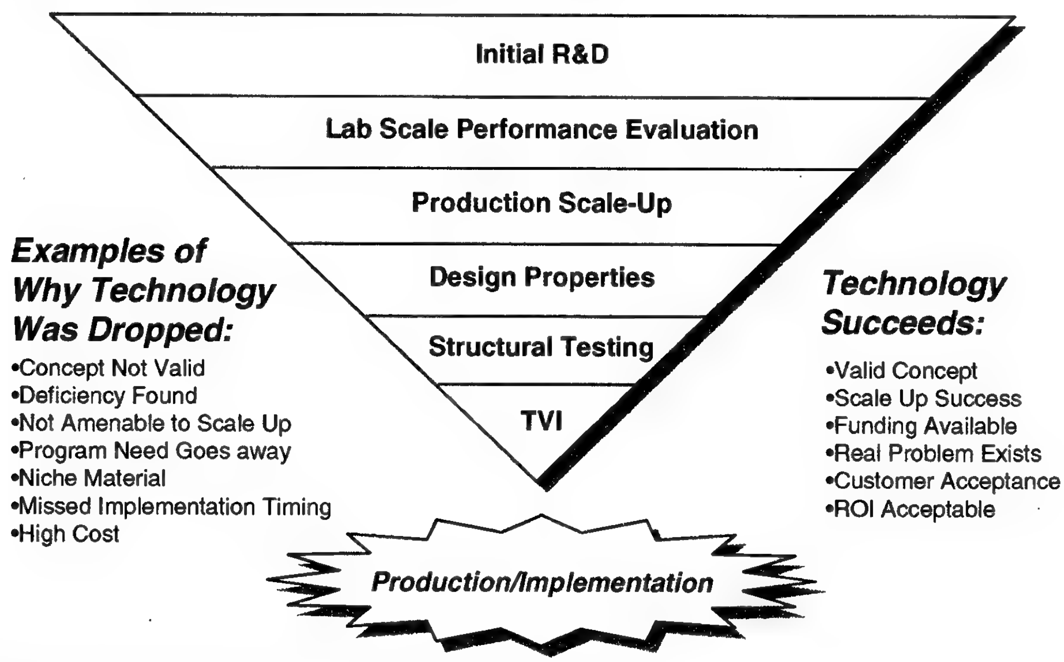


Figure 0-1. Examples of Techology Downselection Criteria

The list of materials implementation challenges seems rather ominous, however, successful implementation of new materials have occurred recently on Lockheed Martin fighter aircraft. These technology insertion opportunities have been an excellent example of industry/government/producer team work. The materials for discussion include a moderate strength Al-Li alloy (2297) and a moderate strength discontinuously reinforced aluminum alloy (6092/17.5p/SiC). Each of these materials are in either full-scale production or are in the process of being qualified for a Lockheed Martin Tactical Aircraft System aircraft.

Discussion

Al-Li Alloy 2297

2297 was initially developed in 1988 under a cooperative research arrangement between Lockheed Martin and McCook Metals, LLC (formerly Reynolds Metals). The impetus for the development was to produce a thick section, reduced density material that had the strength, thermal stability, fracture toughness, isotropy, and corrosion resistance of 2124 plate alloys up to 6 inches. While organic composites have eclipsed metals in two dimensional loading applications such as fighter aircraft skins, acceptable composites for three dimensional loading has proven to be a difficult challenge. Therefore, investing in new metallic structure for highly loaded bulkheads seemed a promising area of research and development. Following several years of alloy development, coupon testing, corrosion testing, and scale-up activities, structural testing was conducted on the main landing gear bulkhead for the F-16 Block 25. Figure 2 shows the outstanding spectrum fatigue behavior demonstrated in the full scale test articles.

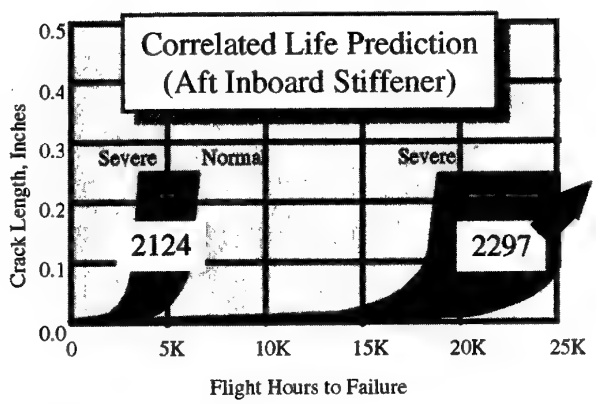


Figure 0-2. F-16 Bulkhead Spectrum Fatigue Results for 2297 Plate

Another key aspect of this testing was to demonstrate that the material behavior was congruent with current fatigue life prediction methodologies. As shown in Figure 3, the coupon testing and component testing resulted in higher stress allowables for the 2297 material compared to even higher strength alloys like 7050-T7451.

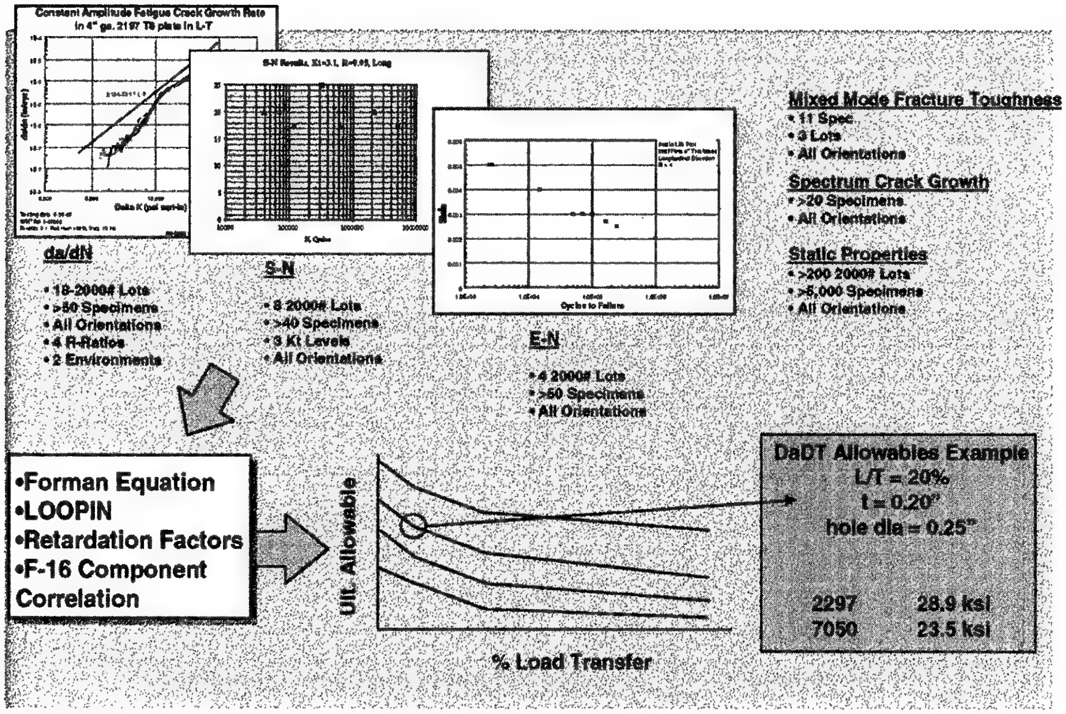


Figure 0-3. 2297 Demonstrated Predicable Fatigue Performance

In addition to the main landing gear bulkhead component testing, full-scale aircraft strain surveys and flight evaluation was conducted at Hill Air Force Base as shown in Figure 4.

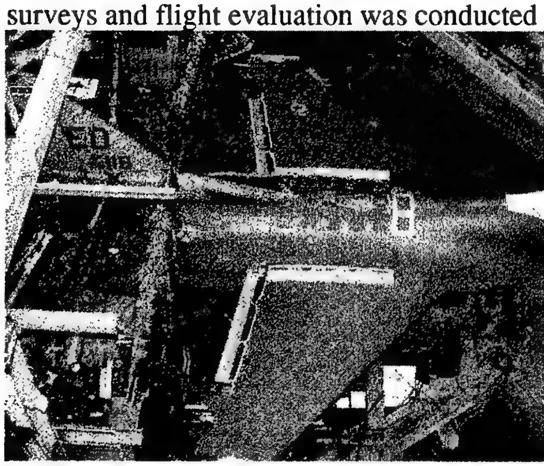


Figure 0-4. Strain Survey of Al-Li Bulkhead at Lockheed Martin TAS

All of the testing was successful and resulted in additional testing for other bulkhead applications such as the aft-most bulkhead on pre-block 40 F-16's as shown in Figure 5. Testing of this bulkhead in a fully-reversed aircraft spectrum resulted in similar 3 to 5X life improvements as shown in Figure 2.

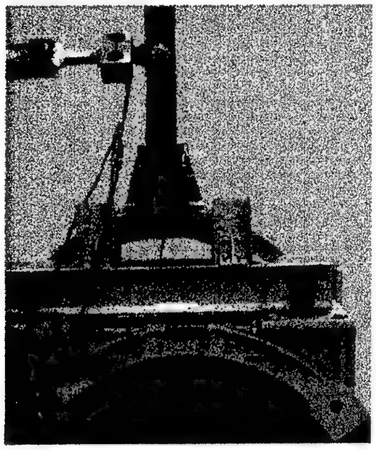


Figure 0-5. Aft Bulkhead Spectrum Fatigue Testing.

As a result of this successful product development program, over 2 million pounds of 2297 plate have been manufactured for use on F-16 spares and new production. ROI discussions will be provided in a later section.

6092/17.5p/SiC Discontinuously Reinforced Aluminum

At the same time as the development of the Al-Li alloy, Lockheed Martin was pursuing high stiffness materials for use in secondary structure applications. DRA materials produced via the powder metallurgy route as shown in Figure 6, offertailorable properties depending on reinforcement type and amount.

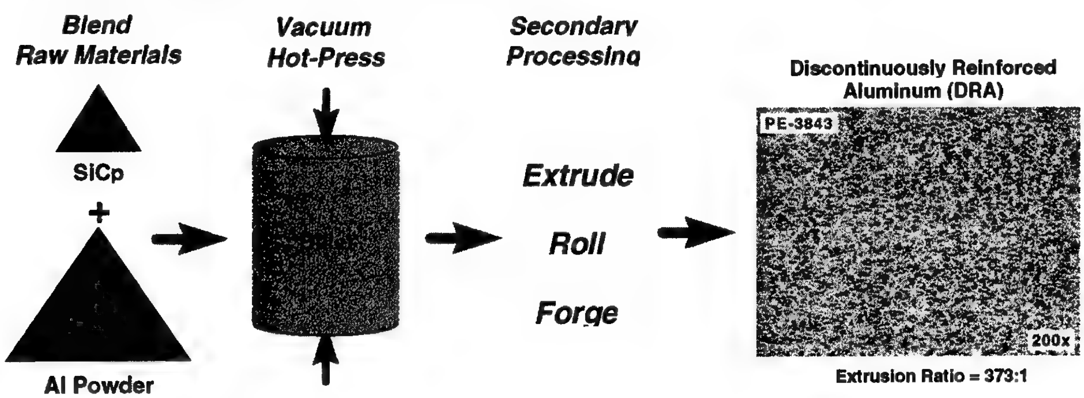
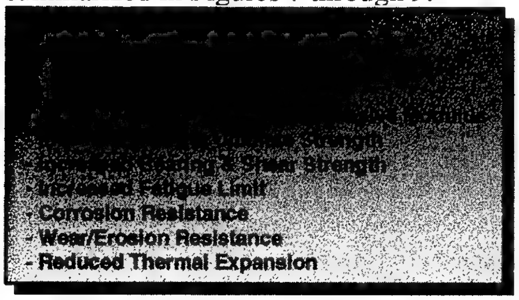
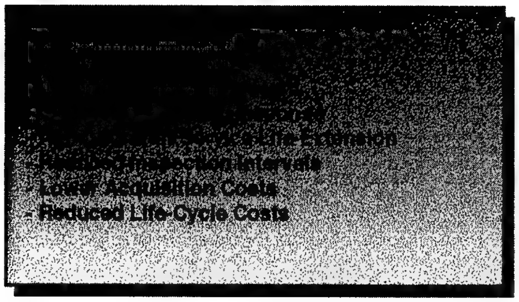


Figure 0-6. DRA Processing Schematic

Other DRA materials have been developed over the years, but the toughness and elongation-to-failure were always a concern for safety-of-flight applications. Lockheed Martin and DWA Aluminum Composites participated in a joint development program to produce a moderate strength, higher toughness material that would meet most secondary structure applications. Following a successful development effort, the 6092 chemistry was selected for scale-up under a "Title III" program under Air Force direction. The Title III program provided allowables testing, fatigue testing, and corrosion testing of production

material to provide an "on-ramp" for production applications. Alloy benefits are summarized in Figures 7 through 9.





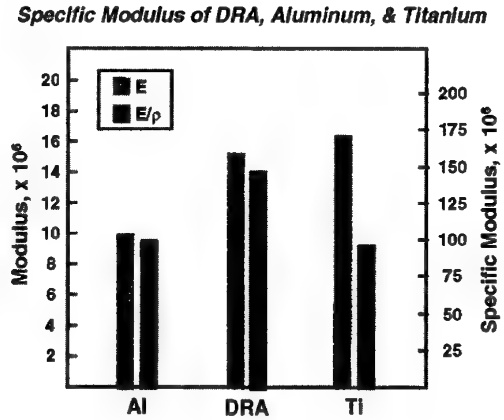


Figure 0-7. DRA Benefits

Material System	6092/SiC/17.5p	6092/SiC/17.5p	6092/SiC/17.5p	6092/SiC/17.5p	6092/SiC/17.5p
Form	Sheet	Sheet	Sheet	Sheet	Sheet
Temper	T-6	T-6	T-6	T-6	T-6
Thickness (in)	.040	.060	.080	.100	.125
Density	$.101\ lb/in^3$	$.101\ lb/in^3$	$.101\ lb/in^3$	$.101\ lb/in^3$	$.101\ lb/in^3$
Basis	Average	Average	Average	Average	Average
F, tu (ksi)					
L	69	67	66	67	67
LT	66	65	65	66	70
F, ty (ksi)					
L	58	58	56	57	57
LT	54	52	52	53	53
F, cy (ksi)					
L	71	66	56	59	62
LT	66	63	56	56	59
F, su (ksi)					
L	44	43	42	42	43
LT	44	43	42	42	42
F, bru (ksi)					
L, (e/D= 1.5) ..	119	118	109	106	107
L, (e/D= 2.0)...	157	150	144	139	140
LT, (e/D= 1.5)..	117	114	108	110	104
LT, (e/D= 2.0)..	156	152	142	139	136
F, bry (ksi)					
L, (e/D= 1.5) ..	117	116	101	100	100
L, (e/D= 2.0)...	147	144	125	124	122
LT, (e/D= 1.5)..	116	112	100	101	95
LT, (e/D= 2.0)..	147	145	123	122	119
e, (percent)					
L	7	8	8	8	8
LT	7	8	8	7	8
E, t (msi)					
L	14.6	14.7	14.7	14.7	14.7
LT	14.7	14.4	14.7	14.7	14.7
E, c (msi)					
L	14.1	14.3	14.3	14.5	14.1
LT	13.9	14.0	14.0	14.5	14.2
CTE, (ppm/°F)					
L	9.3	9.4	9.3	9.1	9.3
LT	9.1	9.2	9.2	9.4	9.1

Figure 0-8. DRA Mechanical Properties

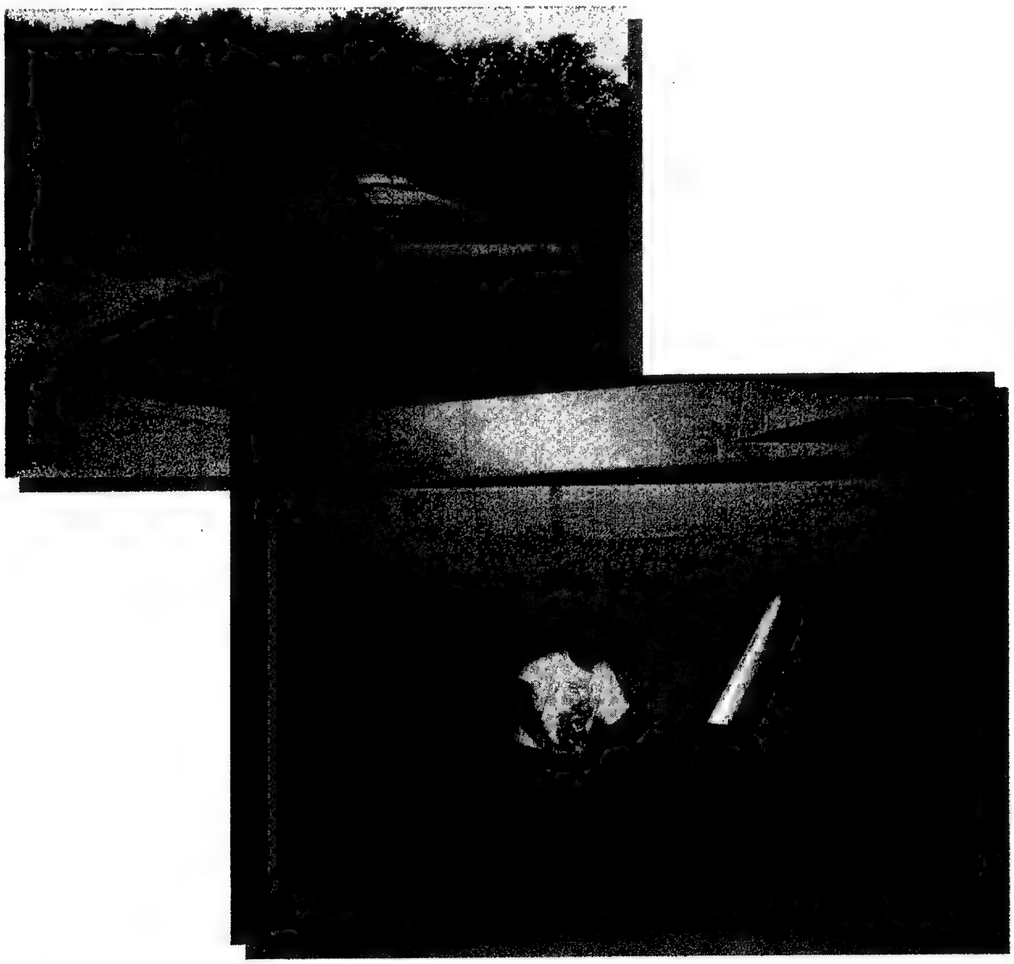


Figure 0-10. Flight Testing of DRA Ventral Fin at RNLAF

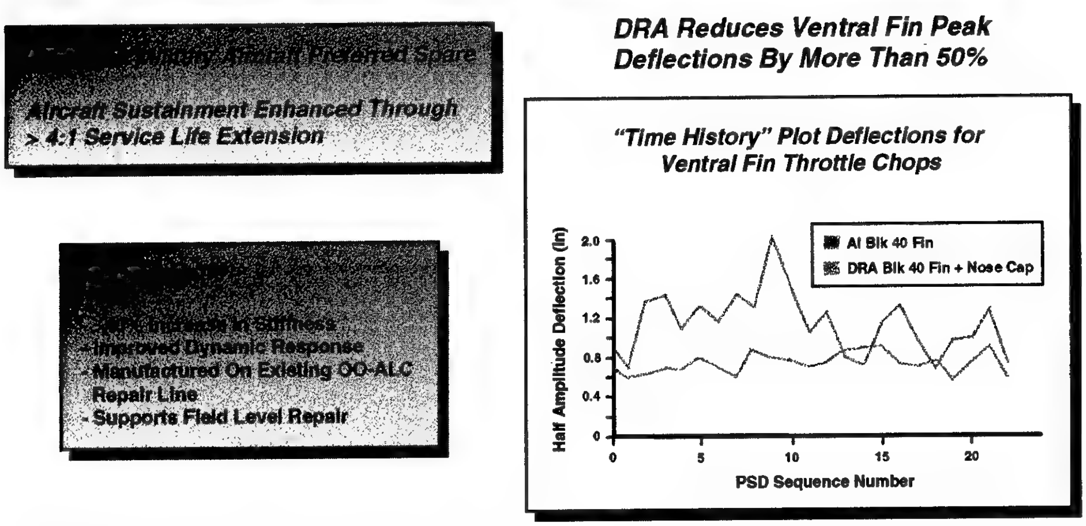


Figure 0-11. DRA Improved Dynamic Response of Ventral Fins

As a result of the successful flight testing on ventral fins, other opportunities to demonstrate the material's higher bearing allowable and higherstiffeness were successfully conducted. F-16 fuel access covers were flight tested inconjuntion with an improved fastening system and demonstrated as much as a 40% reduction in the peak stress levels on the F-16 upper skin as shown in Figure 12.

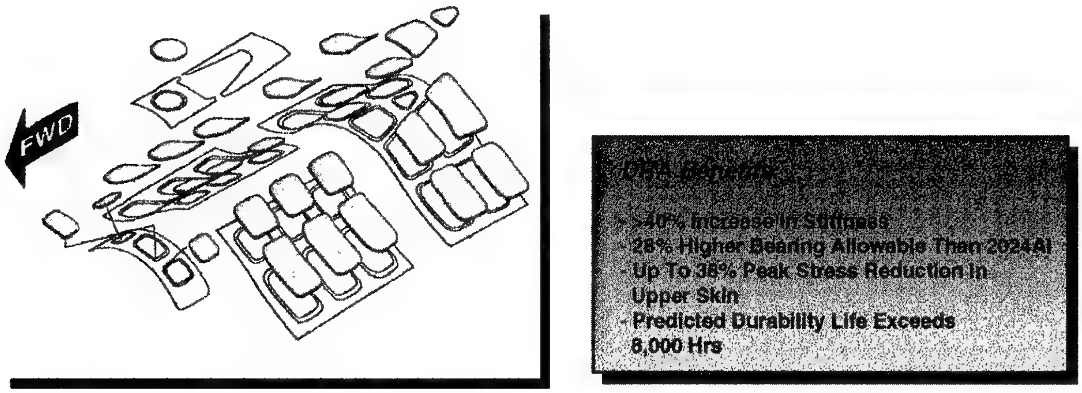


Figure 0-12. DRA Fuel Access Cover Benefits

As a result of these activities, over 200 shipsets of ventral fin skins have been produced at DWA Aluminum Composites for spares applications.

Cost Benefits Analysis

In each of the implementation efforts discussed, the material implementation would not have happened if the return on investment (ROI) was not financially sound. Figures 13-15 give the initial ROI estimates for these applications. The ROI has actually improved since material prices have fallen on both the 2297 and DRA materials once production quantities have been produced.

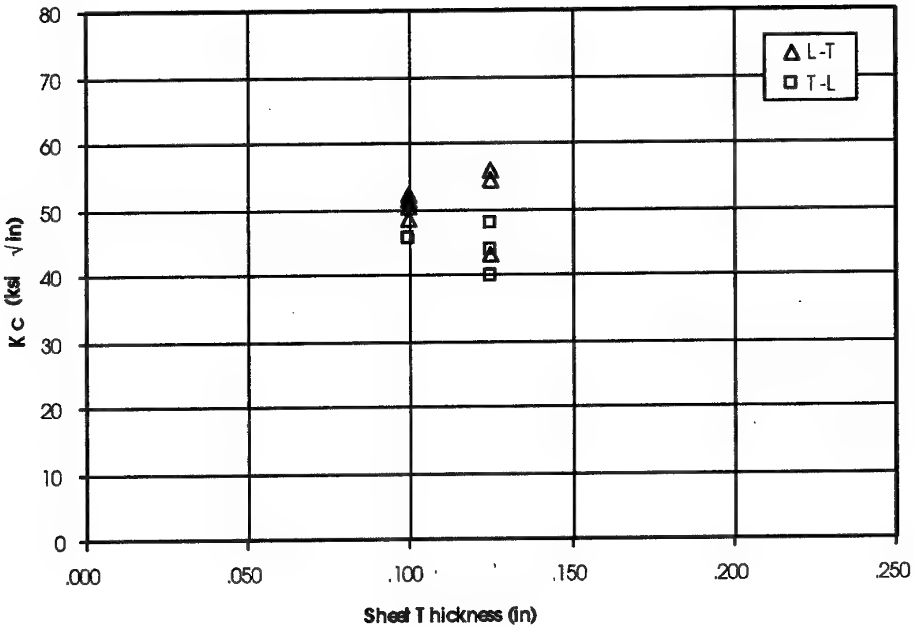


Figure 0-9. DRA Fracture Toughness

As a part of the Title III program, industry participants were selected to provide potential applications for testing and/or full-scale evaluation. LMTAS elected to demonstrate the material on F-16 ventral fin skin manufacturing. The ventral fins are subjected to a highly dynamic environment due to inlet spillage and the various stores arrangements typical in a fighter aircraft. Increasing the stiffness and aerodynamics of the ventral fin skins without increasing the weight, was empirically shown to provide a significant reduction in stress and increase in part life. Flight testing was conducted at the Royal Netherlands Air Force with the support of NLR to document the effect of utilizing DRA skins on a Block 15 aircraft as shown in Figure 10. Flight test results verified the empirical analysis and showed a 50% decrease in in-flight deflections (Figure 11).

Spares Rework Costs at Depot	**Current**	**New Design**
Total Rework/Spares Cost	$3200	$6391

Maintenance Costs Analysis (8000 Hr Service Life)

Inspection	$66,900	$59,820
Maintenance/Replacement Cost	$16,000(5X)	$0 (0X)
Total Costs	$82,900	$59,820
Downtime Caused by Maintenance	2343 Mhrs	745 Mhrs

Projected Savings ⇨ $74,610,000-$53,838,000 = $20,772,000 (900 New Ventral Fins)

Figure 0-13. DRA Ventral Fin Cost Analysis

	2124	2X97
Cost To Replace		
Kit	$15K	$25K
Labor	$35K	$35K
Projected #	3X	1X
Total Fleet Cost (850 USAF)	$127.5M	$51M

Projected Cost Savings = $76.5M

Figure 0-14. 2297 Bulkhead Cost Analysis

CONCLUSIONS

Two successful materials implementation efforts have been described: 2297 Aluminum-Lithium plate products and 6092/17.5p/SiC Discontinuously Reinforced Aluminum. The success of both of these activities were a result of an industry/government/producer team that provided a viable material that was successfully scale-up to production quantities, provided predictable mechanical properties for design, was demonstrated in full-scale test articles, and was successfully applied to aircraft flight test efforts. The ROI for each material provided significant cost avoidance for the end-user.

Technology trends for future Business Jet Airframe

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SUMMARY

Today's aerospace market is extremely tough ; the constant quest for reduced production cost in existing airframes may provide an opportunity for introducing new technologies through re-engineering of structural component.

This paper higlight the approach used at Dassault Aviation for the Falcon business jet family.

Withing the "technologies patchwork", choices and solutions are reviewed and discussed using examples.

1. GENERAL

Several new technologies are under study at Dassault-Aviation for their possible application, first to improve aircraft currently in production (Falcon 900B and 900EX , Falcon 2000 and Falcon 50EX) , and later on future models.

Introduction of new technology is not an objective by itself. Dassault approach (see fig 1) is that a new technology will be introduced only if it provides better value for money for the customer. Or the value side of the balance, safety is always top of the list and never to be compromised ; the other main components of value are dependability, performances (payload, range, speed)and comfort.

- Safety

- Dependability
- Performance
- Comfort

- Fixed & Recurrent costs
- Acquisition cost
- operation & maintenance costs

Value for money balance

fig 1

On the cost side of the balance are acquisition cost , operation and maintenance costs.
To achieve such objective, two of the rules used are :
- One is to maximise the "family effect", i.e. to introduce new technology preferably on parts common to several makes of Falcons, in order to spread development cost on large production runs and also to save on recurring cost by being earlier down the learning curve.

- Another one is to asses and minimise all possible negative effects of a candidate new technology .

2. USE OF METALLIC SUBSTRUCTURE CONCEPT IN RELATION TO COMPOSITE DESIGNS

Organic composites such as carbon / epoxy have strong benefits compared to build up metallic structures:

- potential **weight saving** (up to 30%)

- **corrosion resistance** : carbon / epoxy parts are unaffected by corrosion . This can be a significant cost maintenance saving for example on a composite pressurised fuselage.

- **fatigue resistance** : composite are much stronger in fatigue than metallic materials.

But they have also negative aspects (to be assessed and minimised).

- Low performance for tridimensional loading (ie : fittings)

- **impact damage** susceptibility :

- **repairability** :

- **cost** of the raw material : to recover this initial cost, savings must be sought in "design for manufacturing" using , more integrated design diminishing part count and assembly man-hours , and dedicated new manufacturing techniques for example Resin Transfer Moulding of dry fibre preforms. In this respect use of new metallic substructures concepts are a way of designing innovatives structures with a good integration of metallic and composite component.

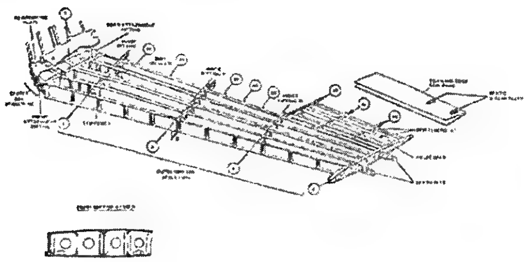
The philosophy is to design simple and robust structures trading one part of the possible weight saving in exchange of good impact damage resistance, easy manufacturing and repairability.

A good example of application of all theses concepts is the re-engineering for the Falcon horizontal tailplane, common to all types of Falcons (F 50, F 900, F 2000).

Existing design is a conventional riveted aluminium structure as shown figure 2 .

Paper presented at the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

Fig 2



New design is shown figure 3.

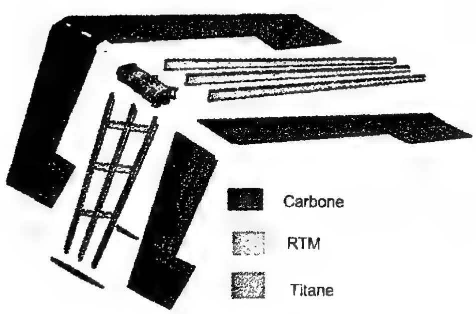


Fig 3

It comprises :

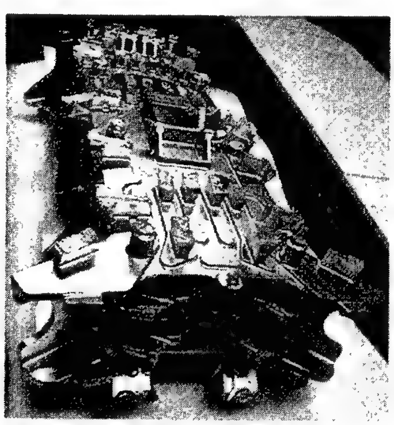
- carbon / epoxy skins, monolithic with some cocured hat stiffeners . The upper surface is a single panel from tip to tip.

- three spars by side , produced by Resin Transfer Molding and integrating some stiffeners and fittings.

- one structural central box consisting in one large titanium (6-4) casting integrating all the attachment fittings (see fig 4)

This part has been produced by investment casting process (see figure 5) .

Fig 4



From the 3D CATIA model, a pattern was first produced by stereolithography, (the use of strereolithography diminishes development time for prototype) a conventional wax model

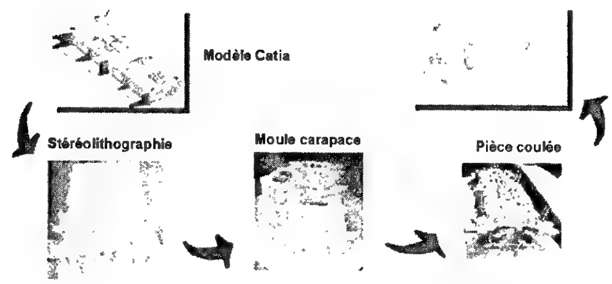


Fig 5

will be used for serie production ; the next step shows the ceramic shell mold ; post casting operations including hot isostatic pressing (HIP) represent a significant portion of cost of producing investment casting .

- leading edge remains the original aluminium one despite trials on low cost titanium SPFDB structure.

The reduction of part count and assemblies is quite obvious permitting significant savings in production cost.

The first horizontal tailplane (prototype) has been manufactured and is being tested with the objective of being certified within this year and being introduced in production line.

3. USE OF METALLIC UNITIZED STRUCTURES.

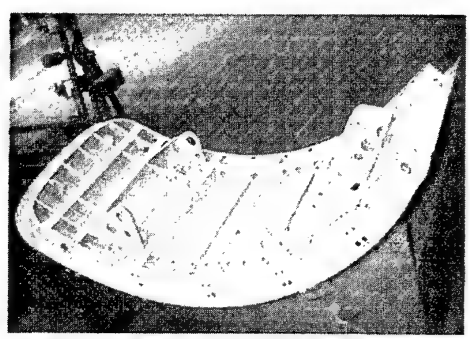
3.1 – Large size casting

For the designer, castings offers numerous advantages compared to conventional build-up structures:
- design freedom
- allows thin walled complex parts
- reduce parts and fasteners count

In contrast the use of large size casting result in difficulties dealing with dimensional tolerances.

Passenger door of Falcon 900 and Falcon 2000 is a conventional build up structure of aluminium sheet metal parts riveted together. (see fig 6)

Fig 6



The idea is there to simplify the structure by replacing all the build up internal structure by a single aluminium cast frame as shown in figure 7.produced by vacuum assisted sand casting.

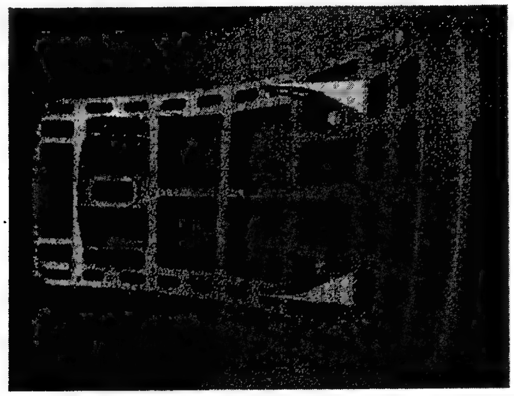


Fig 7

3.2 - High speed machining

The advent of very high speed machining with very high RPM spindle is providing two benefits : first the reduction in milling time, second the possibility to go down to very thin skins without cracking or buckling the part during machining.. That opens the way to cost efficient and weight efficient applications.

Two examples are in development :

- Airbrake (common to Falcons 50, 900, 2000) The initial design (aluminium skin bonded on aluminium honeycomb core) will be replaced by one single integrally stiffened part (see figure 8) with a drastic reduction in part count.

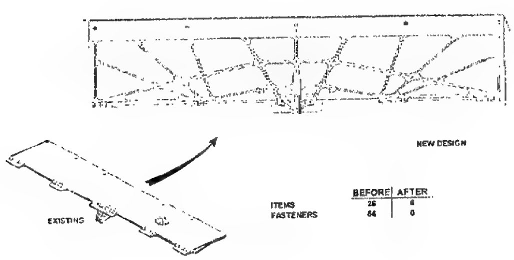
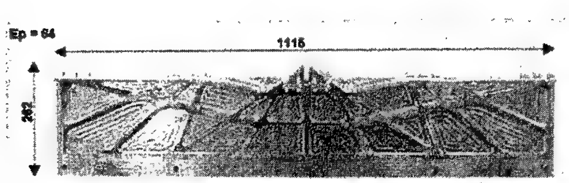


fig8



- Pitch control surfaces (common to Falcons 50, 900, 2000).

In this case, conventional aluminium build up structure will be replaced by a much simpler design using integrally stiffened panels.

4. CONCLUSION

Dassault-Aviation is keen to improve the competitiveness of its products, and for this purpose to introduce new technologies, quipping into mind than all aspects must be properly accounted for, including maintenance.

Paper 3

Question by Mr. Woithe

Have you examples of modification of aircraft parts resulting from fatigue problems in service.

Author's reply

Very few examples at Dassault

- replacement of some wing main spars on some MIRAGE III
- replacement of straker fitting on Mirage F1

Question by Mr. Lincoln

Are casting factors used for design? Has there been any experience with shell inclusions in Ti castings?

Author's reply

There is no casting factor for Titanium 6.4 castings. But in the case of the Falcon horizontal tailplane the titanium fitting has ample static margins because it is replacing aluminium fitting of the same dimensions. All the titanium cast parts are HIP treated and we have had no problem up to now, for example a titanium cast fitting used on the Mirage 2000.

Question by Frank Abdi

What is the margin of safety for fatigue of composite? is traditional I.S. safety considered?

Author's reply

Margin of I.S. is considered only for plane stress. In that case fatigue is never a problem due to the excellent strength of carbon epoxy composites.

Residual strength after impact is often the design device.

Future Aluminium Technologies and their Application to Aircraft Structures

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1. Introduction

Aluminium remains a predominant material for airframes. Carbon fibre composites and titanium alloys have made inroads especially in some military airframes such as Typhoon and Tornado. However with affordability now having equal emphasis to the classical performance requirements in aircraft design, such as speed, range, payload and stealth, aluminium could soon recover some of these applications. Aerospace manufacturers are giving significant attention to developments in the areas of new aluminium materials, low cost manufacturing and unitised structures. The latter is because the cost of producing aircraft is being driven by the cost of assembly which drives production towards fewer, cheaper-to-assemble parts, whilst maintaining close tolerance in manufacture.

2. Emerging trends in aluminium based materials

Currently, the main aluminium alloys used by the UK airframe manufacturers are the high strength Al-Zn alloys 7050, 7150 and 7010 primarily used in strength critical structures and the damage tolerant Al-Cu alloys 2024, 2014 for fatigue critical applications. There are occasional exceptions to this such as superplastic 7475 on Typhoon and high temperature 2618 on Concorde but overall these materials predominate. Thus there is immediate scope for introducing the new higher strength Al-Zn alloys such as 7055 and 7449, the new higher toughness Al-Cu alloys 2024A and 2524 and the new high stiffness, lower density Al-Li alloy 8090 into new airframe applications. These alloys should be able to deliver improved performance fairly readily with the Al-Li alloy particularly being able to offer a 10-15% weight saving providing that the cost premium can be tolerated.(Note 7055, 7449 and 2524 are single sourced which raises commercial issues also.)

The developments on aluminium based materials are progressing on a number of fronts and include strength improvements to Al-Zn and Al-Li alloys, damage tolerance improvements to Al-Cu and Al-Li alloys, high temperature aluminium alloys, high stiffness particulate aluminium matrix composites and the hybrid laminates such as GLARE ™.

2.1.Strength Improvements to Al-Zn and Al-Li alloys

The composition and thermal processing of 7XXX is being further optimised to meet requirements for upper wing skin and extruded stringers. Higher Zn additions (.>8% and similar to 7055 and 7449) have improved attainable compressive strength (~10%) at no penalty to toughness and work continues on the optimisation of these alloys. The improved heat treatment practices such as the retrogression and reageing treatments have improved the stress corrosion cracking and exfoliation resistance compared to 7150-T651.

The alternative to the use of the high strength Al-Zn alloys at very high levels of compressive strength would be the adoption of a high strength Al-Li alloy where the combination of the reduced density and extra stiffness could be taken to offset the need for a very high compressive yield strength. Two UK potential alloy options are available, the first which has stemmed from the Brite-Euram (HSALLI) programme is B13 and the second which has come from MOD research is the isotropic alloy X.

The US Al-Li alloy 2195 (Weldalite) appears to be a cost effective replacement for 2219 in weldable space structures (cryogenic tanks) because its high strength provides superior weight savings. (1) However the material is still heavy relative to normal Al-Li alloys.

2.2.Damage Tolerance Improvements to Al-Cu, Al-Mg and Al-Li alloys

Pechiney (2) have been developing a copper rich 6XXX series alloy that is aimed at replacing 2024-T351. The material offers significantly improved intergranular corrosion resistance and weldability over 2024-T351 at equivalent strength, fracture toughness and fatigue crack growth resistance.

ALCOA (1) have been developing a weldable Al-Mg-Sc alloy intended for use as thin section fuselage components, both sheet and extrusions. The claimed advantages include improved corrosion resistance and lower density when compared with current 2XXX alloys. They have also been developing a modified 2XXX with minimal recrystallisation for thin (and thick) extruded applications to replace 2024/2224. They claim higher strength, minimal machining and improved damage tolerance.

Sumitomo Light Metal Industries, Japan (3) have developed a Al-Mg-Si-Cu alloy with similar properties to 2024-T351 but with the added advantage of good formability and corrosion resistance of 6XXX series alloys. The pressure deck beam of an aircraft was extruded from this alloy. This alloy in a T6511 condition was stronger than 2024 and offered better corrosion resistance. The cost saving of the beam using this extrusion was estimated at 29%.

The UK has been developing dilute Al-Cu-Mg alloys which have demonstrated exceptional damage tolerance in an artificially aged condition and which should provide damage tolerance in a creep aged condition.

The US Al-Li alloy 2097 has good fatigue resistance and is targeted at bulkheads where it should eliminate the need for periodic replacement of 2124, which can develop fatigue cracks..(4) Application to F16 appears to be underway on a case specific basis.

Paper presented at the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

A new UK alloy ALFSOTATS (Aluminium Lithium Fuselage Sheet Optimised for Toughness and Thermal Stability) has been developed which combines dilute composition and a novel heat treatment to produce a material of superior fracture toughness than 2024-T3. (5) The material is targeted at large civil type fuselages.

2.3 The high temperature aluminium alloys

A recent Brite-Euram programme has developed an aluminium alloy for use at 150°C using conventional ingot metallurgy. The new alloy, designated 2650, derived from 2618 through the optimisation of the chemical composition presents improved mechanical properties, mainly toughness and creep, compared to 2618 alloy which up to now has been considered as the reference for elevated temperature applications.

US work on high temperature aluminium alloys for supersonic transport applications has been reported by NASA and Lockheed-Martin. Weldalite alloys RX818 and ML377 and ALCOA alloys C415 and C416 were examined. Better creep properties were exhibited compared to 2618. (6)

A current Brite-Euram programme is developing an aluminium alloy for 200°C plus temperatures using both rapidly solidified aluminium powder and mechanical alloying techniques. The alloy is still in the research phase.

2.4.Spraycast Alloys

Osprey Metals have extended the work done by Alusuisse and Alcan and are now producing a very high strength spraycast 7XXX alloys for commercial evaluation, in billets up to 100kg. (7) The alloy has a Zn content of 11.5% and is believed currently to be the highest strength commercially available 7XXX alloy.

2.5.Mechanical Alloyed Materials

The UK high stiffness particulate metal matrix composites, for example, AMC225xe, produced by mechanical alloying using an Al-Cu alloy base have progressed to small scale commercial production. The material demonstrates gains in specific stiffness approaching 50% and possesses reasonable ductilities.

The mechanically alloyed Al-Mg-Li alloy, AMC 500, which was derived from 5091 has excellent mechanical properties and corrosion resistance and does not require heat treatment. The material offers an 8% density reduction and 15% increase in elastic modulus compared to 7XXX series. The elimination of the final heat treatment is attractive in terms of manufacturing procedures, quench sensitivity, distortion and costs. There are issues around scale up but industrial interest in the material appears substantial.

2.5. The hybrid laminates such as GLARE ™

US work on the fibre reinforced aluminium laminates has continued. The target for second generation GLARE laminates appears to be a civil type fuselage crown where weight savings may be possible with improved durability and damage tolerance.(1)

3. Emerging trends in aluminium manufacturing technology

Manufacturing represents about 95 % of the cost of the airframe, and ways to reduce costs and simplify production are being vigorously pursued. Efforts to reduce the number of fasteners and part count has focused on the design of unitised structures, near-net shape forming, high-speed machining and joining technologies particularly friction stir welding. The cost effective processes such as high strain-rate superplastic forming, creep forming and casting are also receiving considerable attention.

3.1 Unitised integrally stiffened structures

Cost savings of 40 to 50 % may be possible with alternative design approaches that use integral and welded structures to replace the conventional built-up structures.

Boeing, Northrop Grumman, ALCOA, and NASA in the US (8) and British Aerospace, British Aluminium, and Wyman Gordon in the UK are involved in producing integrally stiffened aluminium panels and addressing the crack propagation issues. Two manufacturing techniques are being investigated, machining from solid and extrusion.

3.1.1 High Speed Machining from thick plate

Whilst numerically controlled machinery is a well established process, the advent of high speed machining to produce structural components from thick aluminium plates offers a number of advantages. These include reduced make-span time, consistent quality and the ability to machine thin walled components.

Historically, the balance of properties such as strength, damage tolerance, corrosion resistance and low residual stress required for aerospace applications were difficult to achieve in thick gauges where slow quench rates and low deformations dominate. However, ultra thick plate materials greater than 150 mm are currently under consideration for large components.

Pechiney have proposed an alloy designated 7040 with lower magnesium and copper levels to reduce quench sensitivity in thick sections.(9) Century Aluminium are offering 7050 plate up to 200 mm thick with a class A ultrasonic standard and mechanical properties only marginally reduced from those of 150 mm.(10) ALCOA are also developing thick 7050-T74 material .

BF Goodrich has developed the 'Grid-Lock' structural system in which a cellular structure is produced rather cleverly from machined, interlocking components.(11)

3.1.2. Extrusion

It is possible to extrude wide panels with integral stiffeners and this technology has been applied to transport aircraft in the former Soviet Union where the facilities to produce such extrusions exist.

Both Boeing and NASA are considering extruded integral stiffened panels as opposed to riveted aluminium skin and stringer construction or integral machined thick plate. Furthermore the two companies are also involved in joint

ventures with The Welding Institute (TWI)) and anticipate using a friction stir welded structure in space launch vehicles in the near future.

Closer tolerances in the extrusions are being investigated to reduce costs by minimising the extra operations involved with fitting mating parts.

3.2 Joining

Technologies to join stiffeners to the sheet metal components, extruded section to extruded section and produce the large integrally stiffened structures are being actively investigated. Recent progress has resulted in an increase of confidence in the potential application of welding to primary structures.

Welding technologies that produce joints without significant heat input allow high fractions of the base metal strength to be retained. Laser welding and friction stir welding are examples of such technologies. Laser welding can save both cost and weight, while improving corrosion resistance while FSW results in higher weld ductilities than laser welds.

3.3 Superplastic Forming (SPF)

The superplastic forming of complex shapes including perhaps integral structures is appealing as it is a "single shot" process that achieves net shape forming. A main problem has been the low forming rates for the aerospace materials.

Recently high strain rate superplastic forming has been demonstrated in a number of aluminium based materials at strain rates greater than $10^{-2}s^{-1}$. This was achieved by grain refinement (<3μm) either conventionally or through processes such as mechanical alloying or equal channel angular pressing.

ALCOA, for instance, is developing a new superplastically formable (SPF) sheet which offers weight savings by parts consolidation. The new 7XXX alloy containing scandium is said to combine a high forming rate with the high strength of 7475-T6.(1)

NASA has also reported work on integrally stiffened aluminium panels made by SPF with cycle times down to 4 minutes and using a high temperature adhesive to form joints.

SKY Aluminium (Japan) have developed a method of producing integrally stiffened aluminium structures by roll bonding and superplastic forming. The sheets are clad over certain areas and hot roll bonded to form a layered sheet structure. A small amount of gas generant (used in automobile air bags) is placed into the unbonded areas between the sheets. The roll bonded clad sheets can be heated in a die and gas will be spontaneously produced to expand the structure to the desired mould shape with an internal honeycomb structure.(12)

3.4. Creep Age Forming.

The advantages of creep age forming are that conventional materials can be used and residual stresses are lower. The process can not normally be applied to lower wing skins in 2XXX alloys used in the T351 condition, i.e. stretched and naturally aged. There would be strong interest in using this process for lower wing skins as they have considerably more form than upper skins and various programmes are active in this area. The overriding factor is whether lower wing skins can be formed by this process whilst retaining sufficient damage tolerance.

3.5 Shape Castings

Castings offer the potential for significant cost and mass reduction compared to built-up fabricated structures. In particular, casting technology developments are making it possible to reduce post-casting operations such as machining and assembly. Greater shape flexibility is also possible and the reduction in fasteners due to reduced parts count reduces mass. Whilst castings currently have limited uses on airframe structured, they may find applications on severe curvature skins.

Castings for secondary structures have displaced riveted assemblies, for example on Typhoon. Enabling technologies for castings as primary structures either already exist or could be developed. In general, castings will find their application in components and assemblies of high complexities

4. The exploitation of the new technologies

The selection and exploitation of the new aluminium materials and manufacturing technologies will be a key issue in the competition between aircraft manufacturers, in terms of production costs, performance and enhanced structural integrity. Substantial cost savings in comparison with polymer composites and titanium structures are envisaged. The new technologies coupled with integration of the design and manufacturing process, improving relationships between the primes and their suppliers, the digitisation of the factories will improve the affordability of the new airframes .

The application of these new technologies into older airframes, however, is less clear cut. UK fleet sizes are small, project budgets are small and consequently the qualification and certification costs for the new technologies will tend to prevent their application to the older airframes.

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RRA Heat Treatment of Large Al 7075-T6 Components

by

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ABSTRACT

Retrogression and re-aging (RRA) is a heat treatment process performed on the aluminum alloy 7075 in the T6xxx temper condition to improve its resistance to corrosion, while at the same time maintaining the high strength levels required for aircraft structural applications. For large extruded or forged parts, we have determined that the most practical process involves retrogression at 195°C for 40 minutes, followed by rapid cooling and full re-aging at 120°C for 24 hours. After an RRA treatment of a large extrusion (a three-metre section from a CC-130 sloping longeron), we measured a shrinkage of approximately 0.015%, with minimal distortion damage. There is a small loss of strength, e.g. the RRA yield strength is typically 515 MPa compared to 530 MPa for the same material in the T-6 condition. The corrosion resistance measured both by exfoliation and stress corrosion cracking are significantly better than for the T-6 condition and approach that for the over-aged T-73 condition. Furthermore, the fatigue resistance and fracture toughness of RRA treated material are both within the scatter bands for the T6 condition. For many thick section extrusions and forgings, rework specifications allow for the removal of up to 10% of the material thickness to remove service-exposed corrosion damage (after which the part must be replaced). Hence, the small penalty in strength experienced after the RRA treatment is more than compensated for by improved corrosion resistance, which can eliminate the need to remove corroded material.

INTRODUCTION

The aluminum alloy 7075 in the peak aged heat treatment condition T6xxx (subsequently referred to as T6) has been widely used for structural applications in many aircraft designed in the 50's and 60's. Many of these aircraft are still flying and several models, including the CC-130, are still in production. Corrosion damage is often the reason why 7075-T6 components are replaced in older aircraft, and hence, new parts in this alloy are still available. When improved corrosion resistance is required, alloy 7075 is often used in the over-aged temper T73, but in comparison to 7075-T6, there is a strength penalty of 10 to 15 % which precludes use of the T73 temper when high strength is required. An alternative is to perform the retrogression and re-aging (RRA) heat treatment on parts purchased in the T6 condition. This process was first reported by Cina [1] and over the past twenty years, has been shown to improve corrosion resistance to levels approaching those of the T73 condition while maintaining the strength at levels at or slightly below the T6 condition [2-14]. The first phase of the present work on the RRA treatment of sections of a sloping longeron from the CC-130 (Hercules) aircraft has already been described in a limited NRC report [15], and in two papers [16,17] which cover a background review and detailed results. Some of that work is also included in this paper for completeness together with further tests that have recently been completed.

The CC-130 sloping longeron is a long (9 m) primary structural component in the fuselage of the aircraft. It is roll formed and extruded to the T651 condition, machined and delivered with a coat of epoxy primer and zinc/chromate paint. During the current research programme, we have prepared two open literature papers which describe a range of RRA treatments that were applied to pieces of a longeron (taken out of service due to corrosion damage) [16] and also to new extruded angle material [17]. Results of tensile, fatigue, exfoliation and stress corrosion cracking tests were reported. While some treatments gave better corrosion resistance and some gave better tensile strengths, the most practical RRA treatment for parts of thickness 8.5 mm and above is retrogression at 195 +/-2 °C for 40 minutes followed by re-aging at 120 °C for 24 hours. For parts 4 to 8.5 mm thick a retrogression of 195+/-2 °C for 30 to 35 minutes should be adequate. Rapid cooling after retrogression is required to limit the growth of strengthening precipitates such as $MgZn_2$ which develop during the treatment. Water or glycol quenches are

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preferred for maximum cooling efficiency which leads to maximum strength, but forced air cooling with jets of compressed air is just about as effective and in some circumstances, much more convenient.

The stability of the various phases and microstructural reactions occurring during RRA processing, particularly with respect to resistance to stress corrosion cracking, have been described by Thompson et al [2]. Danh et al. [3] showed that the microstructural processes leading to the enhancement of properties of 7075-T6, consist of:

1) partial dissolution of the GP zones
2) formation and growth of η' particles, and
3) coarsening of grain boundary precipitates, which are primarily η particles.

Wallace et al.[4,5] reported that there are three basic phenomena associated with the increase in corrosion resistance of 7X75-T6 after the RRA treatment:

1) The dislocation density in the RRA treated 7X75 is much lower than that in the T6 condition. This has also been observed by Cina [18]
2) The grain boundary precipitate size and spacing are increased during the RRA treatment and become more effective as sites for the coalescence of hydrogen.
3) After the RRA treatment, the alloy contains a high volume fraction of η' with small amounts of η precipitates. GP Zones may also be present [6]. The η' particles are said to be responsible for the strength of the alloy while the η particles located at grain boundaries are responsible for the alloy's corrosion resistance [2]. Therefore, as the volume fraction of these grain boundary precipitates increases, so does the resistance of the alloy to SCC [7]

EXPERIMENTAL WORK

The experimental work was performed on alloy 7075-T6xxx in four forms with appropriate designations LN, EA, EC and SP as follows:

LN: sections of a service-exposed sloping longeron from the CC130 aircraft, extruded, variable section thickness 8 mm to 25 mm, T6511 (much of the work on this material has already been published - for complete details see reference [16]).

EA: straight lengths of new extruded angle, 101 x 76 x 7.8 mm thick, T6511 (some of the work on this material has been published in reference [17])

EC: straight lengths of new extruded channel, 101 mm wide, 6.3 mm thick, T6511

SP: a stepped bar of different thickness along the length, machined from a 25.4 mm thick plate, 7075-T651.

7075-T73 plate material was tested in some cases to provide comparable results.

In the total programme, a number of variables in RRA processing have been studied [15-17]: *note - not all variables are discussed in this paper.*

(i) method of heating (oil bath, electric furnace)

(ii) temperature of retrogression (180 °C to 220 °C, but only temperatures of 190, 195 and 200 °C were reported in [16] as other temperatures did not provide adequate tensile strengths)

(iii) time of retrogression (seconds to hours depending on temperature, but 30 to 60 minute treatments were selected for further study)

(iv) time of post retrogression aging treatment (0, 6, 12, 18 and 24 hours) [17].

All measuring equipment conformed to AMS 2750 *Pyrometry* and furnace characteristics conformed to AMS 2770 *Heat Treatment, Wrought Aluminum Alloy Parts.*

Before 7075-T6 structural parts can be replaced in service with 7075-RRA treated parts, it is important to determine that the RRA material conforms to the mechanical property requirements of MIL-HDBK-5. Many of the tests described below are listed in MIL-HDBK-5; other tests were included to determine metallurgical characteristics of the RRA material, since it was felt that such information could lead to an understanding of the structural potential if for example the RRA strength was slightly below the MIL-HDBK-5 levels.

The 7075 alloys from several sources used in this study were all subjected to chemical analysis using a Spectrolab Optical Emission Spectrometer (OES). The results were compared against two reference standards, one from NIST and the other from Alcoa, and all compositions were within the allowable limits.

All the results presented below are for tests on the LN, EA, EC or SP material in the as-received (AR) condition and after RRA treatment. Unless otherwise indicated, retrogression treatment is at 195 °C for 40 minutes followed by water quench (W), glycol quench (G) or air quench (A), and re-aging is at 120 °C for 24 hours. An RRA treatment designation **LN-195/40W-12** means the material was from the longeron, received a retrogression at 195°C for 40 minutes, water quenched, followed by aging at 120°C for 12 hours. If a designation such as 195/40W does not include the aging time, it should be assumed that aging is for 24 hours.

The goals of the program were to achieve mechanical properties meeting the requirements of MIL-HDBK-5 for 7075-T6511 extrusions. While it has not been possible to cover all the required properties with the resources available, the results presented do compare favourably with the goals.

TESTING PROCEDURES AND RESULTS

Electrical Conductivity

Electrical conductivity was measured after each heat treatment and compared to the basic T6xxx values. It is

well established that for aluminum alloys, heat treatment conditions which give rise to good corrosion resistance also display higher values of electrical conductivity [19,20], so conductivity is an important monitor. Conductivity was measured using an Autosigma 2000 eddy current meter which utilizes a hand held probe to measure the conductivity in units of %IACS (International Annealed Copper Standard). The conductivity was measured in at least six random locations on each sample and then averaged to obtain a conductivity number for the whole piece. Careful attention was paid to ensure that conductivity measurements were not taken near the edge or at thin sections, so that the electric field of the probe stayed in the metal. The electrical conductivity of 7075-T6 was found to be about 33.5 %IACS, rising to 38 to 39 %IACS after retrogression, and remains at that level after subsequent re-aging. The value for 7075-T73 is required to be 38 %IACS minimum, but is typically about 40 - 42 %IACS. Measurements for several RRA conditions along with the equivalent corrosion results have already been fully reported [16,17]. Long-term (up to 4 months) conductivity stability tests were performed on various RRA treated coupons and it was found that the values were stable over that period [15].

Exfoliation Corrosion

Exfoliation tests, performed according to ASTM G34 *Exfoliation Corrosion Susceptibility in 2XXX and 7XXX Series Aluminum Alloys (EXCO Test),* were used to evaluate the exfoliation corrosion resistance of various conditions of the L and EA material (as-received T6; retrogressed; retrogressed and re-aged; T73) by exposing specimens to an acid saline solution for 48 hours. The EXCO solution consists of NaCl (4.0M), KNO_3 (0.5M), and HNO_3 (0.1M). At least two coupons were prepared for each material/condition tested. The corroded samples were then compared to photographs supplied with the ASTM standard to determine the degree of corrosion. Even though it is a comparative test, the results were reproducible and different observers independently agreed on the results. The EXCO Test classifications in order of decreasing corrosion resistance are N (no appreciable attack), P (pitting), and EXCO (exfoliation corrosion) A, B, C, and D. Various RRA treatments were exposed (190/30W, 190/60G, 195/40G, 200/30W) and little difference was found in the exfoliation performance. The results for the LN material are found in [16] and for the EA material, RRA treatments of 195/40W-24 and 195/40A-24 both gave outstanding exfoliation results, namely that in all tests the rating obtained was P. Not even one coupon exhibited EXCO A or worse.

In summary then, it can be stated that exfoliation resistance after RRA treatments is similar to 7075-T73 i.e. "Pitting", while 7075-T6 performance was much worse, i.e. EXCO B or C.

Salt Spray Tests

Salt spray tests were conducted on EC material in the T6511 and RRA 195/40-24 conditions, according to the requirements of ASTM G85 *Practice for Modified Salt Spray (Fog) Testing.* Preliminary results confirmed the exfoliation results (above) in that various RRA treatments increased resistance to the acidified salt spray as compared to the T6 condition. By exposing different depths to the salt fog it was determined that the RRA treatment was able to provide good corrosion protection through a one cm section thickness, although the pitting corrosion resistance at the outer surface was slightly better. This could be important if fastener holes need to be drilled into a part after treatment.

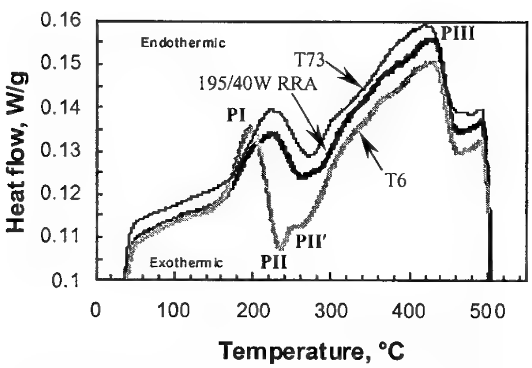
Stress corrosion

Stress corrosion tests on the LN longeron material, were conducted according to ASTM G 38 *Making and Using C-Ring Stress-Corrosion Test Specimens*, machined such that loading was applied in the short transverse direction and the crack formed in the longitudinal (extrusion) direction. The C-ring samples were stressed to 75% of the yield point value of 7075-T6 and then tested according to ASTM G44 *Evaluating Stress Corrosion Cracking Resistance of Metals and Alloys by Alternate Immersion in 3.5 % Sodium Chloride Solution,* i.e. cycling between immersion for 10 minutes and exposure to air for 50 minutes. This process was repeated every hour for 20 days. The samples were checked at the same time every day for cracks and to ensure that no appreciable evaporation had occurred in the test solution. The stress corrosion resistance was rated by the average number of days that a heat treatment could withstand the test before cracking. Cracks were usually of a length of 1 to 2 mm, and located along the back surface of the C-ring specimen.

The results [16] can be summarized as follows: 7075-T73 specimens were able to withstand 20 days without cracking. 7075-T6 specimens all cracked in the first 5 days and the best RRA results were for the 190/50W-24 and190/60W-24 treatments, where all 24 of the specimens survived the full 20-day test. For the 195/40W-24 treatment, only one survived for 20 days, and the other four cracked between day 18 and day 20. In summary, the RRA performance almost matched the T73 results, which is in agreement with earlier SCC results on notched cantilever beam specimens of alloy 7475 [4].

Calorimetry

As mentioned above, the strength of 7075 in various tempers has been ascribed to various precipitates, and the sequencing of precipitation in the microstructure can be determined by differential scanning calorimetry, DSC, [6,11,21-23]. Several T6 and RRA specimens were subjected to DSC (TA Instruments model DSC-2910) by heating a specimen (of known mass) and producing a curve of heat flow (watts per gram) vs. temperature shown in **Figure 1**.



Temper	PI	PII	PII'	PIII
T6511	194	235	262	437
RRA	230	261	281	438
T73	235	277	-	425

Figure 1: DSC curves for 7075-T6, 7075-T73 and 7075-RRA 195/40W materials. The characteristic peak temperatures for each curve are shown in the table.

This curve is of a similar shape and has similar features to the heat capacity/ temperature curve reported by Delasi et al. [21] who summarized the temperature ranges for the primary reactions as follows: **for T6** - *113-217°C = GP zone dissolution, 217-250°C = η' formation + η' dissolution + η formation, 250 - 271°C = growth of η particles, 271-448°C = η dissolution;* **for T73** - *164-245°C = η' dissolution + η growth, 245-442°C = η dissolution.*

In the present case, for the T6 and T73 conditions the temperature ranges were similar, and the peak temperatures PI etc. were very similar to those reported by Baldantoni [6] who studied 220/1W to 220/6W retrogression treatments. The main reason for performing this analysis was to verify the 7075-T6 condition and to determine whether the curve for the 195/40W RRA treatment was similar to that for the T73 condition (i.e. the endothermic dissolution and exothermic precipitation reactions occurred at the same temperatures). Figure 1 clearly shows this to be the case, supporting the conclusions of Baldantoni [6] that RRA treatments (i) complete the dissolution of the GP zones and (ii) - as indicated by the increase in PI temperature - increase the volume fraction of the η'. In the present work, there appeared to be no difference in the DSC response at depths up to one cm below the surface.

Tension Tests

Tension testing was carried out in accordance with ASTM E8 *Tension Testing of Metallic Materials.* Due to material constraints both full size and sub-size specimens were used, and specimens were prepared with the tensile axis in the longitudinal (L) and long transverse (T) directions. A cross-head speed of 2mm/min was used. The 0.2% offset yield strength, ultimate tensile strength and the percent elongation were determined.

For the service-exposed longeron LN material, in the longitudinal direction, tension test results for four different RRA treatments have been reported [16]. In all cases, the ultimate tensile strength (UTS), the yield strength and the elongation results exceeded the minimum requirements of AMS 4169 *Extrusions 7075-T6511.* Note that the AMS 4169 values are the same as the A-basis MIL-HDBK-5 design values [24]. The tensile test results on RRA treated extruded angle material have previously been reported [17]. A summary of those results for the EA as-received (mean of 10 tests) and RRA 195/40W and 195/40A (mean of 20 tests) materials is reproduced in **Table 1** together with MIL-HDBK-5 [24] design values for the same thickness (7.938 mm) extrusions. Although we have not performed enough tests for A- or B-basis statistical

Table 1: MIL-HDBK-5G A-basis mechanical property values for 7075-T6511 extrusions, and test results for the EA 7075-T6511 and EA RRA (195/40W&A-24) materials in the L and T directions.

	Ftu		Fty		Fbru*	Fbry*	e (%)	
	L	T	L	T	L	L	L	T
MIL-HDBK-5 (ksi)	81	78	73	69	146	113	7	
(MPa)	558	538	503	476	1007	779		
EA 7075-T6 (MPa)	613	591	556	530	1080	878	13	13
EA 195/40W (MPa)	583	571	522	515	1079	890	14	15
195/40W-3σ (MPa)	573	569	514	512	1029	803		
EA 195/40A (MPa)	576	568	517	512			14	15
195/40A-3σ (MPa)	568	562	504	506				

* for an edge distance to hole diameter ratio of 2.0

A full definition of the symbols can be found in reference [24], page 1-2, but briefly, Ftu = ultimate tensile stress, Fty = tension yield stress, Fcy = compression yield stress, Fsu = shear stress, Fbru = ultimate bearing stress, Fbry = bearing yield stress, e = elongation.

analysis, we have calculated values for the mean minus three standard deviations for any given condition, which for a small sample is statistically significant. For both the ultimate tensile strength (Fu) and the yield strength (Fty) these values exceeded the A-basis values for both the water quenched and forced-air cooled specimens although water quenching consistently results in higher strength.

Hardness

Although hardness is not considered to be the most reliable indication of the level of heat treat response, Rockwell B hardness was measured both prior to and after each treatment. For the various material sources, the average HRB values for the 7075-T6 condition, the retrogression condition and the RRA (195/40-24) condition respectively were: LN=91,82,90 EA=94,88,92 EC=93,83,90. Rockwell B hardness was also taken on a 19 mm thick section of one of the longeron parts at equal intervals through the thickness to determine if the treatment's effects had penetrated the whole part. These results showed that HRB was constant across the whole section (90+/-1.5) [15].

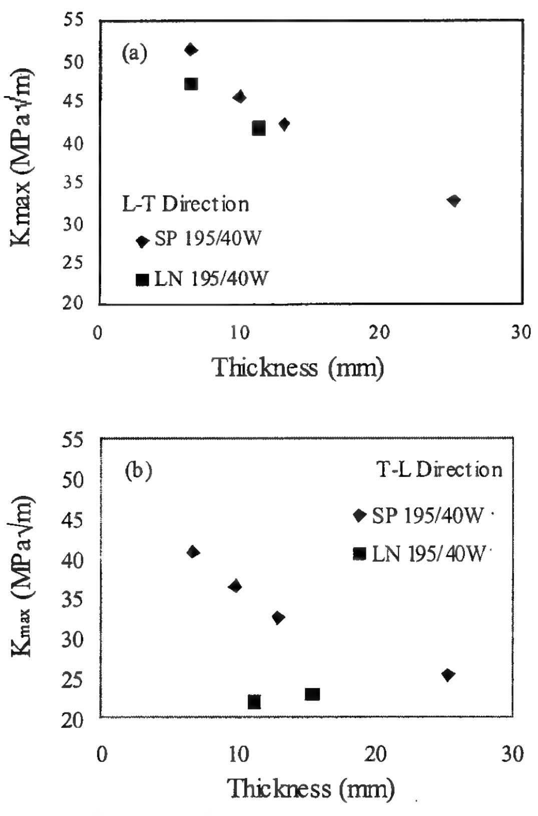


Figure 2: Effect of product form, thickness and specimen orientation on the fracture toughness of RRA treated 7075 aluminum alloy

Fracture Toughness

The material was tested in various product forms and heat-treated conditions as follows:
EA in the -T6511 (as received, AR) condition
EA after 195/40A-24 and 195/40W-24 treatments
LN 7075-T6511 RRA treated 195/40W
SP in the RRA 195/40W condition

Fracture toughness tests were performed in both the L-T and T-L orientations for these materials. All fracture toughness tests were performed in accordance with ASTM E399-*Test Method for Plane-Strain Fracture Toughness of Metallic Materials*. The specimens were compact tension type with characteristic dimension, W=50.8 mm. Loading rates were in the ASTM "slow" regime. Because product forms of various thicknesses were examined, it was not possible to obtain valid plane strain results for the thinner (i.e. most) specimens. Consequently the maximum stress intensity factor K_{max} is reported as the fracture toughness for all material forms and conditions shown in **Figures 2 and 3.**

Figure 2 shows the effect of material thickness on the fracture toughness of the RRA treated materials. In both cases, the fracture toughness in both L-T and T-L directions decreases with increasing thickness. In most cases the measured values are above the typical values for 7075-T6 quoted in the ASM Metals Handbook [25].

Figure 3 shows the effect of product form (EA, SP and LN) and treatment on the relative fracture toughness for specimens of constant (7.8 mm) thickness. Again it is

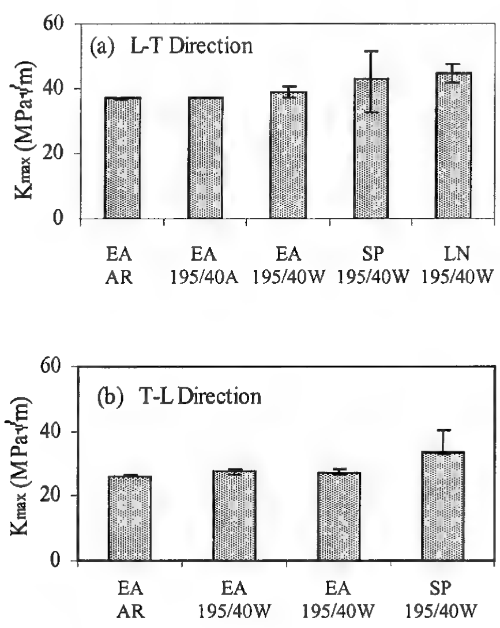


Figure 3: Effect of specimen orientation, quench medium and product form on fracture toughness of RRA treated material.

observed that 195/40W and 195/40A RRA-treated material exhibited fracture toughness at least as good as for the 7075-T6511 temper.

Fatigue

Axial loading (R=0) test The material was tested in axial fatigue using specimens machined from EA in the as-received (AR) T6511 condition. Both 195/40A and 195/40W RRA treatments were examined. In addition, two surface conditions were investigated:

(i) machined and mechanically ground and polished

(ii) as-extruded.

All axial fatigue testing was performed in load control in accordance with ASTM E466 – *Practice for Conducting Force Controlled Constant Amplitude Axial Fatigue Tests of Metallic Materials*. The specimens had a rectangular gauge section that was 12.3 mm x 4 mm for the machined and polished specimens and 12.3 mm x 7.8 mm for the as-extruded surfaces. The data shown in **Figures 4 and 5** collected in this study for lives clustered around 10^7 cycles represent unbroken specimens (i.e., tests interrupted before fatigue failure occurred).

The loading was repeated tension (R=0) with cyclic frequency between 5-30 Hz depending on specimen life. The fatigue life results are plotted in **Figure 4** for the polished specimens. The figure also shows the fatigue data from MIL-HDBK-5G [24] for 7075-T6xx material at R=0. In general, it can be seen that the RRA specimens exhibit slightly lower fatigue strength than the AR specimens but the data are generally within the scatter of -T6xx data. Also, the water quench appears to give slightly better fatigue strength than the air quench.

The data for the as-extruded surfaces are compared to those for polished surfaces in **Figure 5**. As expected the as-extruded specimens all have lower fatigue strengths than the polished specimens at all stress levels, although at high stresses, the differences are not so pronounced. For the as-extruded condition, the RRA results are practically identical to the as-received T-6 results, so neither benefit nor loss has been derived in this respect by the RRA treatment.

Fatigue Crack Growth

Fatigue crack growth data were measured on EC 7075-T6511 as-extruded channel and on EC 195/40W RRA treated material. The specimens used were single edge notch (SEN) type, 76 mm wide and with the original thickness of the channel extrusions cut from the centre of the backs of the channels. The testing was performed with the loading axis in the L direction. All crack growth testing was done in laboratory air at cyclic frequencies of 10-30 Hz following ASTM E647 –*Test Method for Measurement of Fatigue Crack Growth Rates*.

The crack growth results for 7075-T6511 and RRA (EA 195/40W) materials at several stress ratios and maximum stresses are shown in **Figure 6.** Two maximum stress levels were used for the RRA material, 50 MPa and 30 MPa, and stress ratios of R = -0.3, 0.1, and 0.5 were examined. In Figure 6, for the R values of -0.3, ΔK is the positive portion of the range in accordance with ASTM E467.

The data generally show that the RRA material exhibits slightly greater resistance to fatigue crack growth than the –T6511 material under similar loading. This is consistent with the previous observations that the RRA-treated material is more ductile, has a slightly lower yield strength and higher fracture toughness than the 7075–T6511 material.

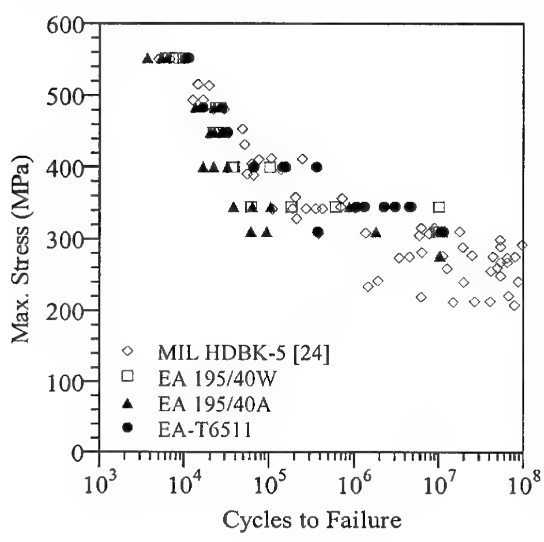


Figure 4: Constant amplitude axial fatigue data at R=0 for 7075-T6511 and after 195/40W and 195/40A RRA treatments.

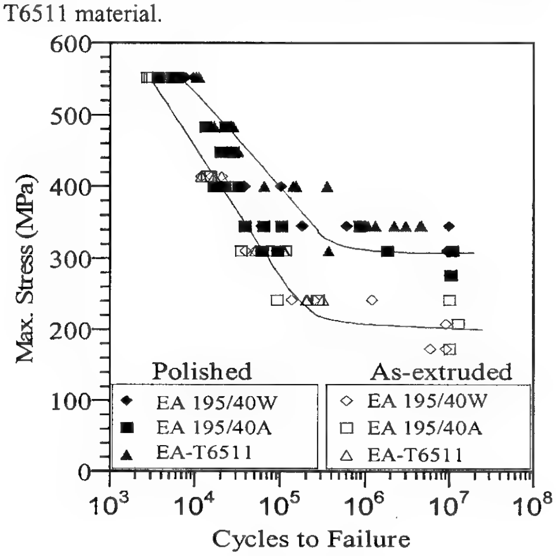


Figure 5: Effect of surface condition on the axial fatigue behaviour of as-received 7075-T6511 and RRA treated materials.

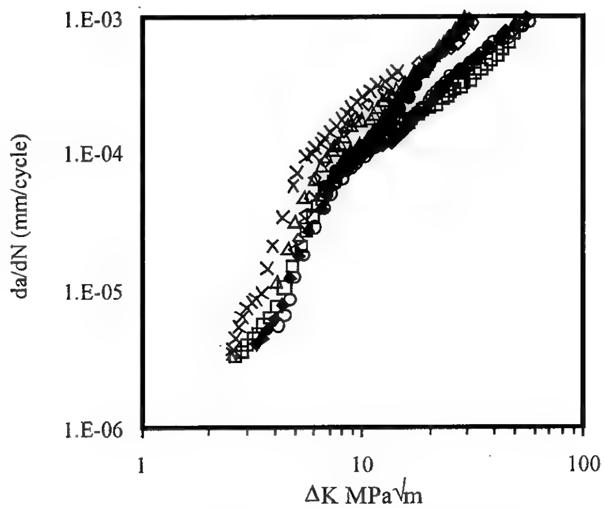


Figure 6: Single edge notch fatigue crack growth data for 7075-T6511 and EA 195/40W RRA treated material.

Corrosion Fatigue

Rotating bending (R=-1) constant amplitude fatigue tests were conducted on EA 7075-T6511 and 197/35W RRA treated materials with and without the presence of a corrosive medium. A cantilever-type rotating bending fatigue machine equipped with a mechanism for dripping EXCO solution (see above for formula) on the gauge section of the specimen during cycling was used. The gauge diameter of the specimens was 7.62 mm and loading frequencies of 30-60 Hz were used depending on the fatigue life.

The results of the fatigue tests with and without corrosion are shown in **Figure 7**. The corrosion test is extremely aggressive, and the results show that both the 7075-T6511 and the RRA-treated samples exhibit fatigue lives on the same curve (or in the same narrow scatter-band) under these conditions (open symbols). This indicates that the RRA treatment has slightly improved the corrosion fatigue properties, since the RRA tests in air fall in the lower portion of the scatter-band (closed symbols in Figure 7 which in turn are similar to the axial fatigue results reported above in **Figure 4**). Unfortunately, it was not possible to include corrosion fatigue studies on 7075-T73 in this programme for comparison. As expected, the corrosion fatigue lives are well below the fatigue lives of specimens tested in laboratory air, especially at lower stress levels.

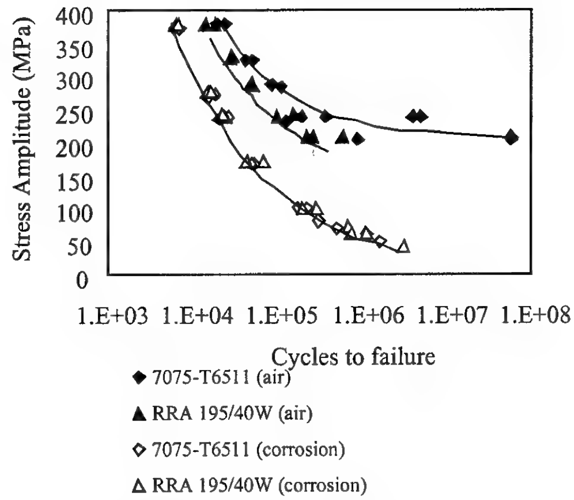


Figure 7: Rotating bending fatigue data for 7075-T6511 and RRA treated material, tested in air and in NaCl solution.

Effect of Pre-Fatigue on RRA treated Materials

Rotating bending fatigue tests were used to determine if the RRA treatment can eliminate or reduce existing fatigue damage. This may be an important consideration for example in choosing whether to perform the RRA treatment on components which have experienced a considerable amount of service exposure, and which might therefore contain fatigue-induced damage. EA specimens were pre-fatigued to 60% of their expected life at several stress levels, then given the 197/35W RRA treatment. The fatigue tests were then continued at the same stress levels. The results were given in reference [17]; at all stress levels, the pre-fatigued specimens fell in the same scatter-band as the T6511 and RRA results, so the RRA treatment had no significant effect on pre-fatigue-induced damage.

Bearing Load

ASTM E 238 *Standard Test Method for Pin-Type Bearing Test of Metallic Materials* is used to calculate the bearing strength of materials under edge loading with a close-fitting cylindrical pin. It is a comparative test used in the design of structures, primarily to determine material performance in the presence of fasteners. The material for the test was EA, both in the as-received T6511 and the RRA 195/40W-24 condition. The bearing load tests were performed on material treated at IAR and tested at SPAR Aerospace on specimens 6 inch (152.4 mm) long, 1.5 inch (38.1 mm) wide and 0.25 inch (6.35 mm) thick. The edge distance (e/D) ratio was 2. Due to specimen thickness and width constraints, a smaller pin than specified in ASTM E238 was used in the tests, which might be expected to result in lower bearing yield (Fbry) and ultimate (Fbru) strengths. However, for both the T6511 (mean for 5 tests; Fbry = 878 +/- 41 MPa, Fbru = 1080 +/- 24 MPa) and RRA (mean for 10 tests;

Fbry = 890 +/- 29 MPa, Fbru = 1079 +/- 17 MPa), the results were comparable to those for 7075-T6 extrusions published in MIL-HDBK-5 [see **Table 1**]. In all the bearing load tests, RRA specimens (with shear-out failures) consistently exhibited about twice the ductility of the T6 specimens, which failed by tensile-cleavage fractures.

Dimensional Stability

Selected parts were analyzed for size and shape both before and after treatment. They were characterized by their profile along one edge and the distances between various fastener holes. A difference in the profile of the part after treatment indicated bending, whereas a difference in the distance between rivet holes indicated shrinkage. The analysis used a Temmis Laser System to identify points on the parts surface with relation to a reference plane. To measure the distance between holes the circumference of the circle was identified with the same system and the center located and projected to a reference plane. Then the distance between the two centers on the reference plane was found. The same procedure was used on each part before and after heat treatment.

The results, which are described in detail in reference [15], can be summarized as follows:
(i) The RRA treatment causes a slight shrinkage - the maximum shrinkage recorded was 0.015%, equivalent to 1.5 mm in a 10 m length;
(ii) The RRA treatment induced a slight bending distortion - the maximum bending distortion recorded over a length of 1.3 m was 0.20 mm, which is minimal and can easily be overcome during installation of a large component.

DISCUSSION

The most important RRA [195/40W-24, and 195/40A-24] results can be summarized as follows:

Corrosion: Exfoliation, salt spray and stress corrosion cracking results for RRA material are always significantly better than for the -T6 condition and approach the performance of the -T73 condition. Several previous studies have come to the same conclusion [2,8,9].

Tension: For the service exposed longeron material, the mean tensile strengths reported in reference [16] for a 195/40G RRA treatment are about 1% above the MIL-HDBK-5 (and AMS 4169) values. The mean values minus one standard deviation (σ) were about 1% below the MIL-HDBK-5 values. However, for new extrusion material (EA) reported herein and in reference [17], 195/40W and 195/40A tensile results for both L and T orientations, for a minimum of 10 tests, indicate that (mean-3σ) values exceeded the MIL-HDBK-5 A-basis values for ultimate tensile strength (Ftu) and yield strength (Fty). There may be several reasons for the better results on the new extrusions - (i) the longeron material, being older, may not have had the (even new) strength of recently manufactured 7075-T6 extrusions; (ii) the service exposure may have weakened the material (e.g. damage incurred in service or during removal) and (iii) the higher scatter in the LN results gives rise to the higher standard deviations.

Although the typical values for as-received T-6 material was always slightly higher (by 3-5 %) than the RRA/W, which in turn was stronger than the RRA/A material, the RRA treatments consistently resulted in higher elongation (14 %) than 7075-T6 material (12 %).

Fracture Toughness Figures 2 and 3: The RRA EA material had the same or slightly higher fracture toughness in the L-T and T-L orientations than the as-received material. Fracture toughness decreased with section thickness.

Fatigue: The fatigue strength of the RRA material was slightly lower than the -T6 material, although the results did fall within the same broad scatter-band as the T6 results published in MIL-HDBK-5. Hence, fatigue should not be a design issue in the selection of the RRA process to improve corrosion resistance. Water quenching after the retrogression step gave better fatigue properties than did air cooling.

Fatigue Crack Growth Rate: the RRA process improved the FCGR which is in agreement with the higher observed ductility so it would be reasonable to assume that the damage tolerance for cracked structures would also be improved.

Physical Characteristics: Since re-aging is a low temperature treatment, distortion is not considered to be a problem, even for large parts. Also, as reported in Reference [16], the RRA treatment does not damage epoxy primer paint either on service-exposed or new parts. Since many new parts are delivered in this condition, it is beneficial that the RRA treatment can be performed without paint removal.

In the present program, over the past four years, a number of different RRA treatments that were performed on various 7075-T6 extrusions have been evaluated and those results have been presented in previous publications [15-17]. It was found that for parts up to 25 mm thickness, the optimum RRA heat treatment was retrogression at 195 °C for 40 minutes, followed by water or glycol quench and aging at 120°C for 24 hours. Similar results have been obtained for 197/35W-24 treatments, but shorter retrogression treatments at higher temperatures, and shorter aging treatments failed to meet the MIL-HDBK-5 requirements for yield strength [17]. Some of the earlier results have been included in this report for the sake of completeness. Results for RRA

treated EA material, together with MIL-HDBK-5 A-basis values (for equivalent thickness extrusions) are presented in **Table 1.** Due to lack of resources, compression and shear tests have not been performed in this programme even though these properties are included in MIL-HDBK-5. However, for all tests where ductility is a factor, (tension, bearing load, fracture toughness) the RRA ductility was higher than that for the -T6 condition, so the RRA results for compression and shear tests would be expected to be favourable.

From Table 1 it can be seen that mean values (determined from 10 to 20 tests), both for the new, EA as-received 7075-T6 material and the RRA treated material all exceed the MIL-HDBK-5 A-basis values by at least 3σ (i.e. 3 standard deviations). Although we have not performed enough tests for an A-basis statistical computation, mean-3σ can be considered to be a conservative approximation. For the service -exposed LN material, the tensile strengths were marginally lower, so RRA treatment for service-exposed material is only recommended for components that are life-limited by corrosion. Note that in order to generate A- and B-basis design allowables for the RRA heat treatment process an extensive test program would have to be undertaken. Statistically valid sample sizes would address variability in product forms (plate, bar, extrusion, forging), grain direction (L, LT) and production lots.

When choosing an optimum RRA heat treatment for a structural component, the tensile strength results are considered the most important as tension values are used as one of the main selection criteria for design purposes. Hence, although the 190/50W & 190/60W RRA treatments gave better SCC results, the tensile results for 195/40W were better. In cases where 7075-T6 components are removed from service because of corrosion damage, then there is a strong case for performing the RRA treatment on the new part prior to installation. It is estimated that the life could be tripled before equivalent corrosion is incurred. Even for fatigue- or tension-loaded critical components, there may be justification for accepting small fatigue and strength penalties of the RRA material if corrosion has been found to be the life limiting factor. Obviously each case will have to be considered individually, but it should be noted here that many component drawings do allow for the removal of corrosion damage by grinding (sometimes up to 10 % of the section thickness in thick forgings and extrusions), so 7075-T6 components which are thus designated, are particularly strong candidates for the RRA treatment.

Some thought has been given to performing the RRA treatment on service exposed components and returning them to service. This practice is not recommended except as a temporary measure if new parts are not available.

It is hoped that in the near future, an RRA-treated (small) component will be installed on the CC-130 aircraft during a routine overhaul. The selected part will be corrosion-prone and will be readily and regularly inspected for a period of three years, during which time, in the -T6511 condition, it might be expected to show signs of corrosion damage.

CONCLUSIONS

1. For 7075-T6 extrusions of up to 25 mm thickness, the optimum RRA heat treatment is retrogression at 195 °C for 40 minutes, water quench and re-age at 120 °C for 24 hours. Compared to the 7075-T6 condition, the RRA treatment:
 - Significantly improves the corrosion resistance.
 - Improves ductility, fracture toughness, and fatigue crack growth rates and slightly improves bearing load strengths and possibly corrosion fatigue resistance.
 - Slightly decreases (by approx. 2%) the tensile strength, tensile yield strength and possibly also the fatigue strength.
2. Hence the RRA process is ideally suited for 7075-T6 parts and components which are usually replaced due to corrosion problems. Where tensile strength or fatigue resistance are critical, water quenching after retrogression is required, but when corrosion is the main cause for concern, parts may be forced-air cooled after retrogression either for convenience or if a slight shrinkage cannot be tolerated.
3. There is a slight penalty in strength when RRA treatments are given to (older) service-exposed material, so the treatment in this case should be limited to those parts which are not strength or fatigue critical.

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Paper 7

Question by Mr. Frank ABDI

Introduction of heat treatment might change grain size?

Author's reply

We have not found any change in the grain structure after performing the RRA treatment

Question by Frank Abdi

What is the composition of new precipitate material, and what temperature is it formed?

Author's reply

We have not performed any analysis of the fine precipitate particles which form during the RRA treatment. The scanning calorimetric results indicate that the precipitate reactions are similar to those occurring during the T73 heat treatment.

Question by I.G. Palmer

What is the current patent situation with regard to RRA treatment?

Author's reply

Alcoa hold a number of US patents for a "three step aging process" for 7xxx series alloys which closely resembles the RRA treatment. Early NRC work is cited as a reference for prior work. I do not know of any patent coverage in Europe.

Question by I.G. Palmer

Are the problems in applying RRA treatments to thick sections in 7075-T6 due to quench sensitivity effect? is the process more effective in the newer, less quench sensitive 7xxx alloys?

Author's reply

We are only working with 7075 - T6 as this is the alloy of concern in aging aircraft. To date we have only examined parts of section thickness up to 19mm thick, for which the RRA process works. well. We are planning to do work on thicker sections in the near future.

Improved Durability Aluminium Alloys for Airframe Structures

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ABSTRACT

The longevity of an airframe depends on its design, service conditions, and the durability and damage tolerance (DADT) of its constituent materials. Critical materials properties include corrosion resistance, fatigue resistance, fatigue crack growth resistance, as well as toughness. The present paper focuses on the metallurgical parameters that govern these properties and on recent developments aimed at improving the DADT of airframe alloys.

Resistance to structural corrosion of aluminium alloys for airframes depends on the nature and number density of the metastable hardening precipitates as well as on local alloying element concentration profiles, in particular in the vicinity of grain boundaries. The underlying corrosion mechanisms are discussed in the context of the development of two new corrosion-resistant products for use as upper wing skin panels (7449-T7951) or fuselage skin panels (6056-T78) respectively.

The fatigue resistance of structures is mainly governed by the fatigue resistance of joints, which is strongly related to technological factors (machining tolerances, riveting practice, etc.). Materials testing to simulate this behaviour is performed on open-hole specimens. Overall fatigue resistance of thick plate, for example, depends on ingot quality and rolling practices. Better understanding of these parameters has enabled the development of aerospace-quality 7010, 7050 and 7040 plate at thicknesses of up to 215 mm.

Finally, longevity of airframe structures depends on the materials' resistance to fatigue crack propagation and their fracture toughness. The metallurgical parameters that govern these properties are discussed and illustrated by recent results on fuselage and lower wing skin 2xxx alloys.

1. INTRODUCTION

Commercial and military aircraft are increasingly required to have longer service lives. As a result, considerable effort has been put into the study of ageing aircraft and the life extension of existing aircraft. However, it is clear that the airframes of the future need to be designed in the expectation of longer service lifetimes. Moreover, there is a cost-motivated drive towards increasing inspection intervals on new aircraft. As a result, there is a need for improvements in the durability and damage tolerance of airframe alloys.

A schematic illustration of the way in which materials properties can impact on the lifetime of a DADT-dominated metallic airframe part is presented in figure 1. Resistance to corrosion phenomena and fatigue life affect the time until initiation of a detectable flaw. Subsequently, fatigue crack growth rate, in more or less aggressive environments according to service conditions, dominates the growth of this crack. Finally, the residual strength of the structure, and thus the critical crack length, depends amongst other factors on the toughness of the base material.

In this paper, the metallurgical principles underlying these critical properties are discussed and illustrated with examples of recent or ongoing alloy development work (see AA compositions of alloys mentioned in table 1).

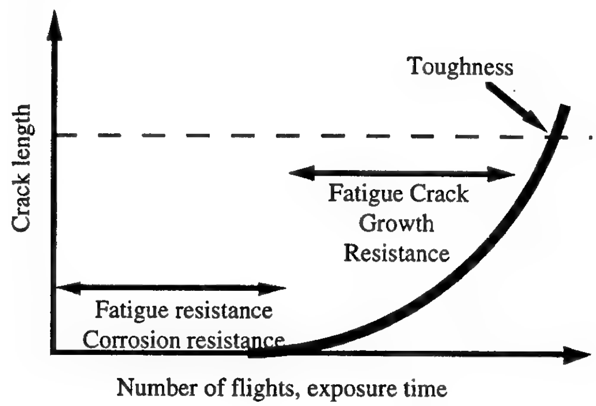


Figure 1. Schematic illustration of the influence of materials properties on the lifetime of an airframe structure.

Table 1. AA registered compositions of the recently developed alloys cited in this paper.

Alloy	Si	Fe	Cu	Mn	Mg	Cr	Zn	Zr
6056	0.7 1.3	0.50	0.5 1.1	0.4 1.0	0.6 1.2	0.05	0.1 0.7	<0.20 (Zr+ Ti)
7449	0.12	0.15	1.4 2.1	0.20	1.8 2.7	0.05	7.5 8.7	<0.25 (Zr+ Ti)
2024A	0.15	0.20	3.7 4.5	0.15 0.80	1.2 1.5	0.10	0.25	0.05

Paper prepared for the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

2. CORROSION RESISTANCE OF AIRFRAME ALLOYS

Localised corrosion phenomena such as intergranular corrosion (IGC) and exfoliation corrosion can limit lifetimes of airframe structures, in particular by providing flaws or even pre-cracks for subsequent crack propagation. Sensitivity to IGC itself is a purely electrochemical phenomenon (see e.g.[1], [2]) determined by the existence of a microgalvanic couple between a grain boundary feature constituting a continuous anodic path with respect to a cathodic matrix (e.g. a continuous film of Al_3Mg_2 precipitation in Al-Mg alloys or a copper-depleted zone near the grain boundary for Al-Cu alloys). It is generally recognised ([3], [4]) that sensitivity to exfoliation corrosion almost invariably results from the conjunction of an aligned grain structure and sensitivity to IGC.

In order to avoid sensitivity to IGC of 2xxx or copper-rich 7xxx alloys it is necessary to devise a processing schedule that precludes the development of and/or reduces solute concentration profiles in the vicinity of grain boundaries. This can be achieved by limiting heterogeneous precipitation at grain boundaries during quenching or subsequent ageing (see figures 2 and 3 for application to the sensitivity to exfoliation corrosion of 7449 extrusions, and also [5]) and/or by overaging to equalise solute concentrations between grain boundaries and the matrix. The relative ease of desensitisation with ageing also depends on a suitable choice of alloy composition; for copper-rich 6xxx alloys, for example, there is an optimum ratio of Si:Mg:Cu that enables effective desensitisation to IGC for a minimal reduction in strength. This effect has been exploited in the development of 6056-T78 for fuselage skin and stringers, which combines a suitable composition window with a targeted overage (see e.g. [6], [7]). Another example of an optimised overageing treatment is the development of 7449-T7951 that allows a 10% increase in compressive yield strength with respect to 7150-T651 with significantly improved corrosion resistance (see figure 4 and [8]).

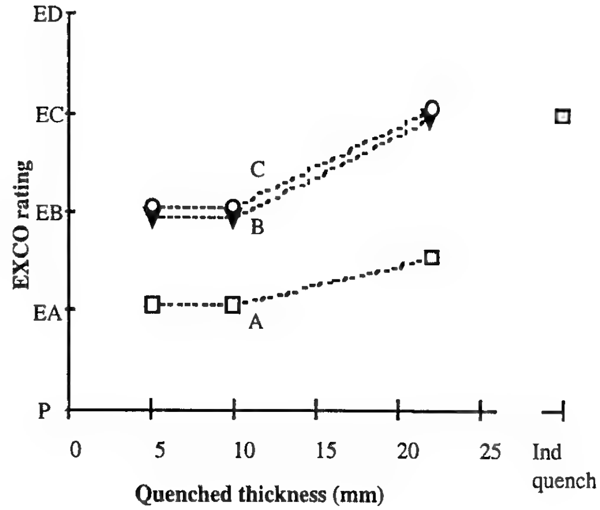


Figure 2. Influence of quench rate on exfoliation corrosion resistance of heavy-gauge 7449 extrusion (approx 1" flange thickness) in three near-peak tempers. Note the improvement in EXCO rating (P (best) -> ED (worst)) with increasing quench rate (decreasing quenched part thickness). This can be attributed to a concomitant reduction in heterogeneous precipitation at the grain boundaries during quench (see text). Temper A corresponds to a slight under-age at 120°C, tempers B and C respectively to a near peak age and a slight overage in an ageing cycle including a higher temperature step at 150°C.

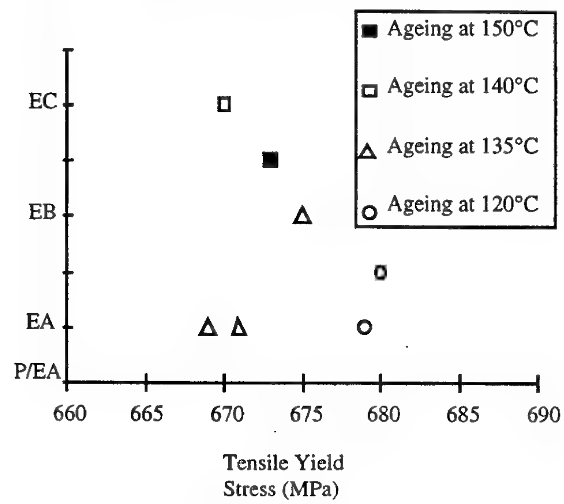
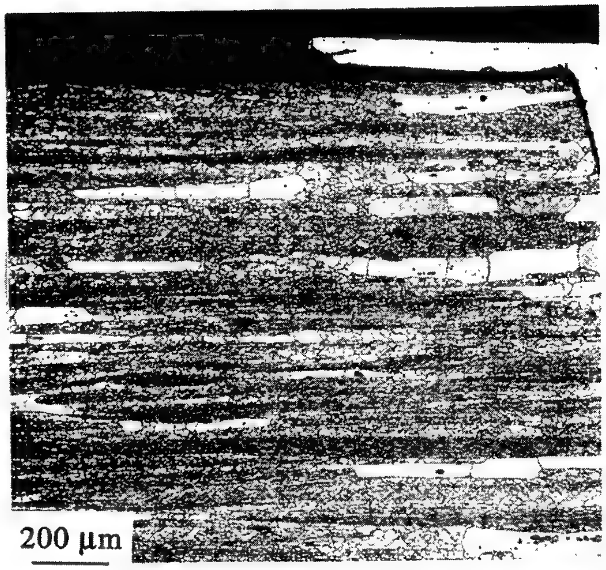
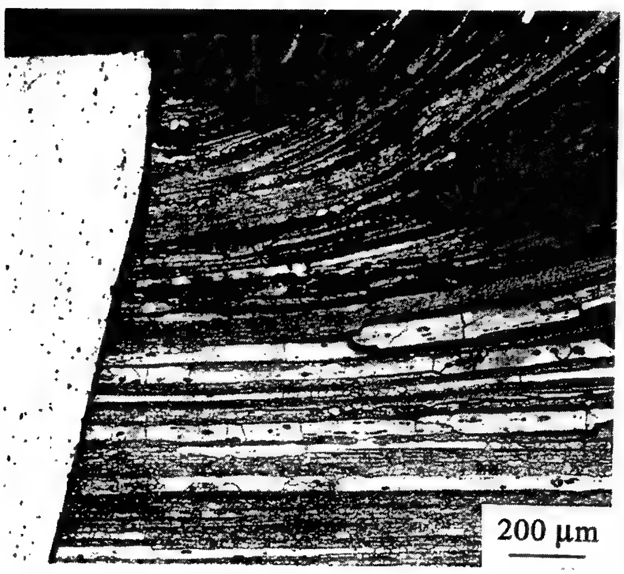


Figure 3. Exfoliation corrosion as a function of tensile yield stress for 7449 heavy gauge extrusion (sampled at th/10) aged to peak strength with ageing cycles with different peak temperatures. EXCO rating (P (best) -> ED (worst)) is degraded with higher temperature ageing, which can be attributed to an increased propensity for grain boundary precipitation (see e.g. Kirman[9]) and concomitant local solute depletion with ageing temperature. Note that the ageing treatment at 150°C (only) was preceded by a first step age at 120°C.



(a)



(b)

Figure 4. Comparison of corrosion in the vicinity of a 2117-T4 rivet in (a) 7449-T7951 and (b) 7150-T651 plate after 5 months exposure in a sea coast atmosphere at Salin de Giraud, on the Mediterranean coast (chromic etch of L-ST cross-section). The exposed surfaces correspond to the mid-planes of 38 mm (7449-T7951) and 25 mm plate (7150-T651). Note the considerably reduced sensitivity to exfoliation corrosion of the overaged 7449-T7951, which nevertheless has a compressive yield stress 10% higher than that of 7150-T651.

3. FATIGUE RESISTANCE

The fatigue resistance of structures is mainly governed by the fatigue resistance of joints, which itself is strongly related to technological factors. However, the material itself plays a role and fatigue tests on specimens containing one or two holes are aiming at evaluating the intrinsic materials fatigue resistance in its position in an assembly.

The fatigue resistance of these hole-containing specimens is related to the distribution of coarse intermetallic precipitates (their number density and more markedly their size) and to the yield stress of the material, both playing a role in the initiation stage[10]. It also depends on the fatigue crack growth behaviour of the material, which is discussed later.

Beside this main aspect of fatigue, it is necessary to assess the fatigue resistance of the bulk material[11]. Improved smooth specimen (K_t= 1) fatigue resistance of thick plate depends on ingot quality and rolling practices. Better understanding of these parameters has enabled the development of aerospace-quality 7010, 7050 and 7040 plate at thicknesses of up to 215 mm. An example of the effect of process modifications on a given alloy is given in figure 5. A model relating fatigue life and defect size was used to evaluate the different production practices on the fatigue resistance[12]. Below a certain pore size, the intermetallic precipitates become the predominant feature for initiation of fatigue cracks.

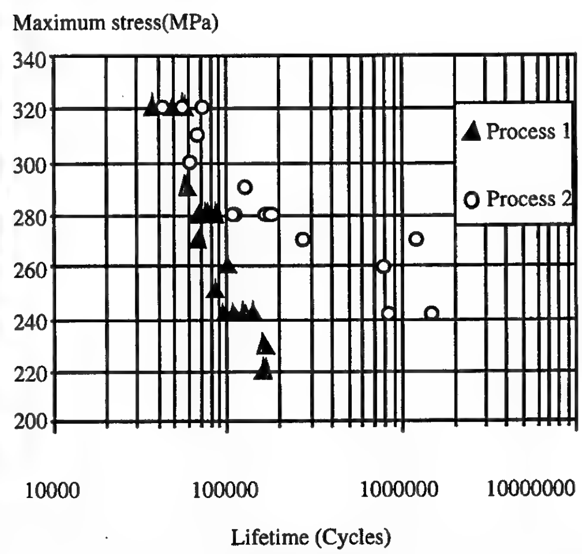


Figure 5. Effect of optimised process parameters on fatigue resistance of thick plate (7010-T7451 200 mm plate, R=0.1, Kt = 1).

4. FATIGUE CRACK GROWTH RESISTANCE

The design of many commercial aircraft lower wing and fuselage skins is limited by fatigue crack growth resistance and toughness. Currently, the alloy of choice for these applications is 2024-T351 or T3, but there is increasing demand for the development of an alloy with an improved balance of properties for these applications. Given the complex nature of the property compromise required for these applications, the replacement of 2024 is not a simple matter. As part of an R&D effort to develop an improved alloy for damage tolerance dominated structures, we have undertaken a background study on the microstructural paramaters that govern FCGR of 2024-type alloys. Some of the results of this work are presented in the following paragraphs.

The parameters mentioned in the literature as important with respect to crack propagation rates of aluminium alloys include (see e.g. [13, 14]):

- a beneficial effect of slip reversibility, which is affected by the nature of the hardening precipitates and dispersoids, and by the substructures;

- a detrimental effect of coarse intermetallic particles, particularly for higher ΔK values;

- the cyclic behaviour of the material.

In addition, closure and environmental effects must also be considered in the understanding of this behaviour.

In order to quantify some of the above mentioned effects, we have characterised the fatigue crack growth behaviour of model plates of 2024-type alloys in CCT specimens (sampled at quarter thickness, W = 200 mm, B = 5 mm, L-T orientation) at R = 0.05 or R = 0.5. Plates with a variety of dispersoid number densities (high, medium, low in figures 6 and 7) and extreme grain structures (largely unrecrystallised to completely recrystallised, and completely recrystallised with coarse grains) were processed and then characterised. With respect to their FCGR performance, the following conclusions can be drawn:

- the observed FCG rate increases with increasing dispersoid number density at R = 0.05 (see figure 6);

- FCG rate at R = 0.05 is lowest for the low dispersoid number density, coarse recrystallised grain material;

- there is little or no influence of recrystallisation rate on the FCGR of the high dispersoid density material.

In the tests at R = 0.5, the differences observed at lower R-ratio between coarse grain low dispersoid content plate, classical low dispersoid density plate, and medium dispersoid density plate are no longer significant.

These differences in FCGR are reflected in differences in crack propagation paths (see figures 8 and 9). Crack paths are extremely perturbed in the recrystallised, low dispersoid density plates, with locally well-defined planar crack surfaces and significant secondary cracking. For the higher dispersoid density plates (see e.g. figure 9) the crack paths are macroscopically very flat, though very locally perturbed. These characteristic differences in crack path are observed at both R-ratios.

These different paths can be attributed to the relative homogeneity of deformation as a function of dispersoid density (see e.g. [13, 14]). Relatively low dispersoid density allows planar slip over large distances, and thus crack paths that are very perturbed on roughly the scale of the grain size. Higher dispersoid densities homogenise slip, and thus favour macroscopically flat crack surfaces. The fact that the difference in crack propagation path is retained at R=0.5 but not the improvement in crack propagation rate of the low dispersoid density plate over the other 2024 plates implies that at least part of the difference in FCGR at low R-ratio is attributable to crack closure effects that are suppressed at the higher R-ratio.

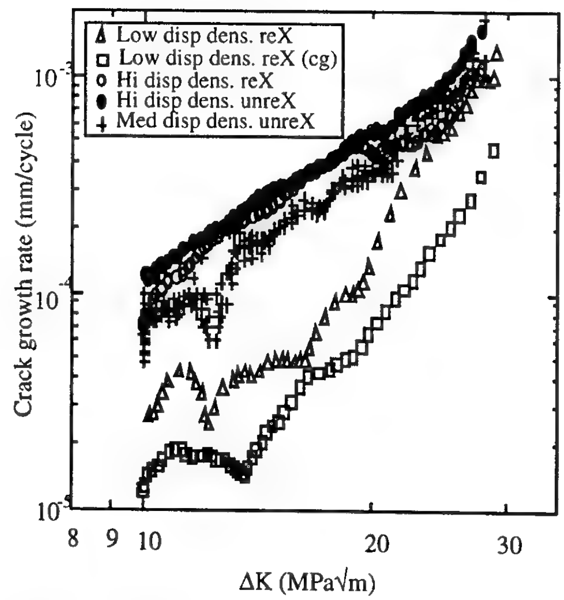


Figure 6. Comparison of fatigue crack growth rates at R = 0.05 as a function of crack growth rate for different 2024-type alloys (CCT specimens, sampled from 40 mm plate, different composition-processing variants). See text for discussion.

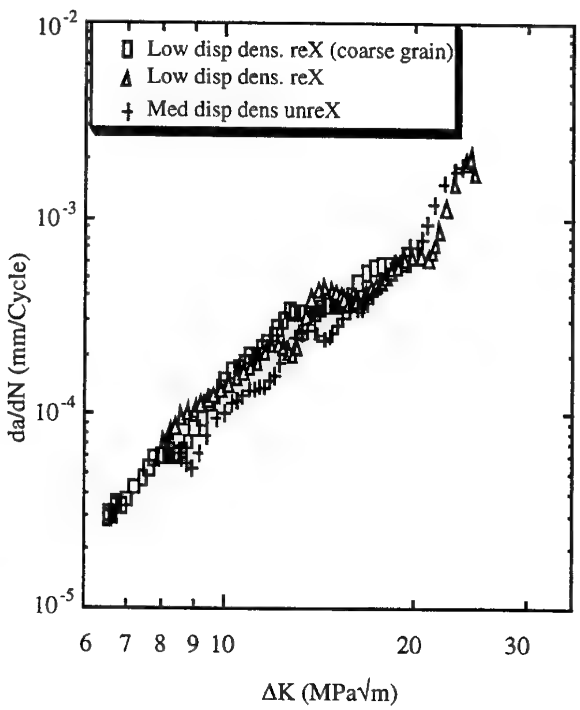


Figure 7. Comparison of fatigue crack growth rates at R = 0.5 as a function of crack growth rate for different 2024-type alloys (CCT specimens, different composition-processing variants). See text for discussion.

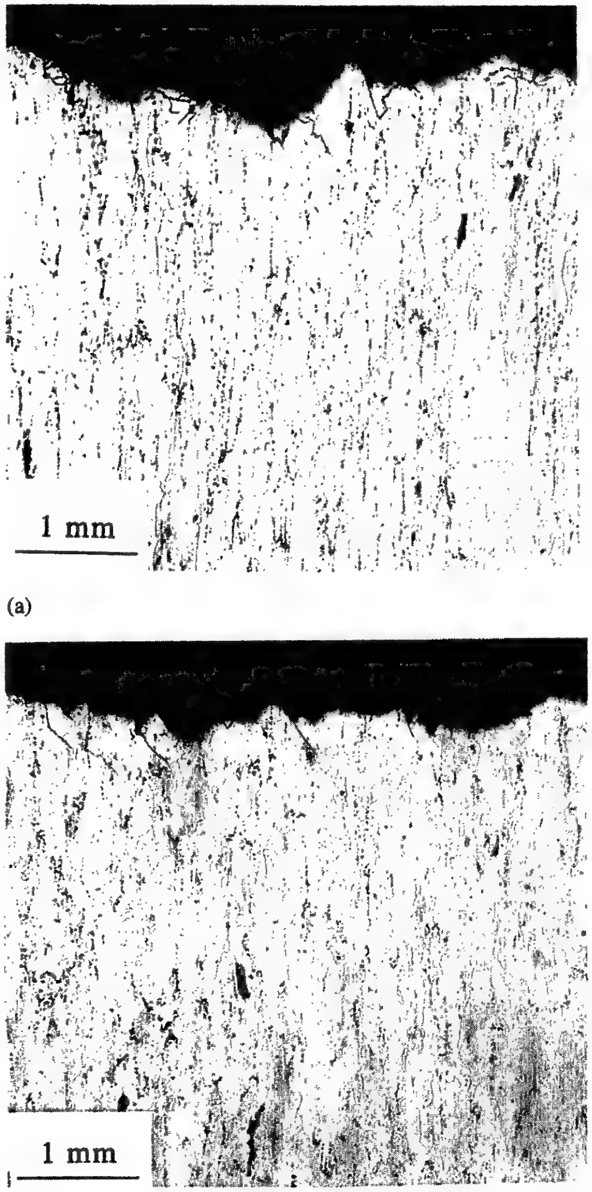


Figure 8. Crack paths at ΔK = 15 MPa$\sqrt{m}$ in the low dispersoid density plates (L-T sections, after chromic etch): (a) recrystallised, coarse grain size, (b) recrystallised, classical grain size.

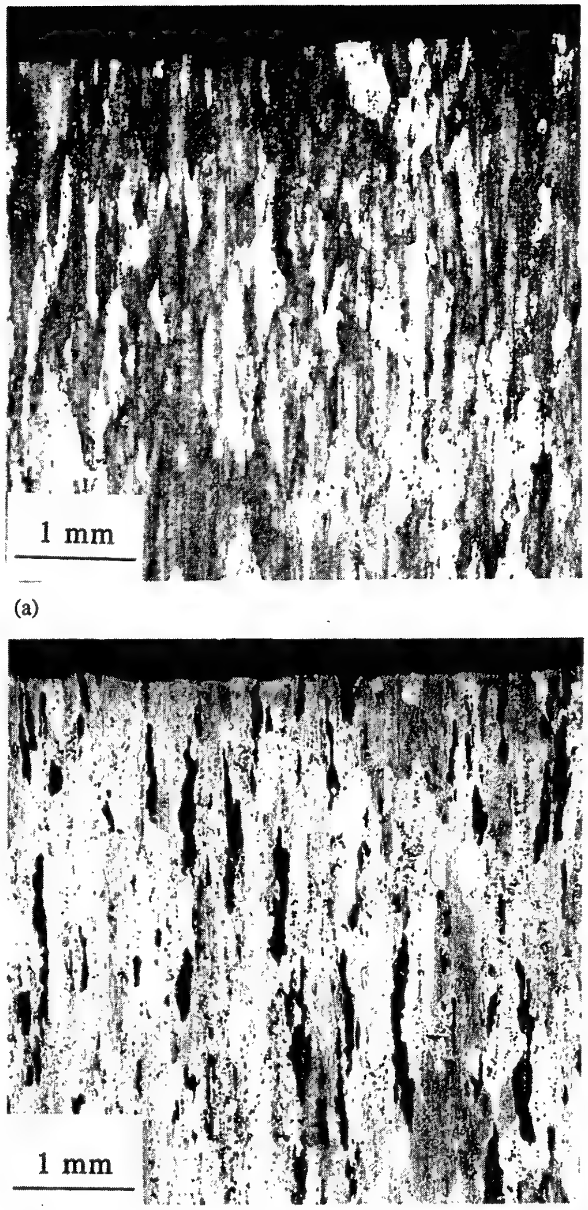


Figure 9. Crack paths at ΔK = 15 MPa$\sqrt{m}$ in the high dispersoid density plates (L-T sections, after chromic etch): (a) largely unrecrystallised (b) largely recrystallised.

The same line of reasoning can also account for the improvement in FCGR at R = 0.05 for the coarse grain material. Coarser grains allow of slip over a larger scale, and thus render more marked the closure effects observed for the low dispersoid density plates. For the higher dispersoid density, the effect of modifying the grain structure is not detectable because slip is already homogenised by the dispersoids on a smaller microstructural scale.

These data, combined with similar results on other key properties for lower wing skin and fuselage applications, are currently being exploited in the design of a new lower wing skin product.

5. TOUGHNESS

The microstructural parameters that determine the toughness of aluminium alloys are well-described in the literature (e.g. [15], [16]). A recent example of the application of some of these principles is the development of improved toughness versions of 2024-T3 fuselage sheet. Improvements in processing schedule and composition that enable a reduction in the volume fraction of coarse intermetallic particles and dispersoids in 2024-type alloys have been identified. The corresponding improvements in materials properties are indicated in table 2.

Table 2. Typical properties of 2024A-T3 sheet compared with conventional 2024-T3 sheet (data for 1.6 - 3.2 mm sheet)

Property	Direction	2024A-T3	2024-T3
TYS (MPa)	L	340	350
	TL	300	320
UTS (MPa)	L	455	465
	TL	440	455
El (%)	L	22	19
	TL	24	18
K_{C0} / K_C(MPa√m)*	L-T	105 / 155	93 / 140
	T-L	100 / 150	90 / 130

* tested at W = 760 mm, $2a_0$ = 133 mm, calculated for W = 400 mm.

6. CONCLUSIONS

The development of improved alloys for DADT-dominated applications requires detailed understanding of microstructure-property relationships, some of which have been discussed and illustrated above. However, significant weight gains on new airframes will necessitate the simultaneous improvement of several such properties, each one not necessarily varying in the same sense with a given microstructural (process) modification. Clearly, close collaboration with airframe designers in order to quantify key property goals. Moreover, quantified metallurgical modelling of static and DADT properties of industrial wrought alloys is therefore a key tool in providing efficient responses to these property goals. Research efforts to this end are underway[17]. Application of the results of this R&D on an industrial scale requires both an understanding of the possibilities and limitations of industrial processing and continuous improvements in the mastery of the processing schedule.

ACKNOWLEDGEMENTS

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Studies on long term durability of aluminum airframe structure made by affordable process

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SUMMARY

Affordability is one of the most important problems of today's development of airframe especially for aluminum alloy application. Some new aluminum alloys and improved processes are being applied to production cost reduction, and tests related to long term durability of applied structures are also carried out.

In this report, our recent studies of following three affordable process methods to aluminum alloy structures are introduced, these are outline of process studies and their merits, and mechanical properties, fatigue properties and corrosion resistance.

(1)Application of new 6000 series alloy of high formability
(2)Application of premium precision casting
(3)Application of superplastic forming

1. INTRODUCTION

Aluminum alloys and composites are competiting severely in the field of airframe structural material. Development of composites increases its speed now, and these are now applied to primary structures. However application of aluminum alloys seems to be maintained as major structural materials because progressive improvement of aluminum alloy and its process have been also made.

Weight reduction and cost reduction are pursued by development of new alloys and improved forming process.
Trend of improved aluminum forming process is to change complicated assembled structure to monolithic structure. These are achieved by large scale high speed machining, large panel precision forming, and relatively large size complicated precision casting and superplastic forming. Application of high cold form material may considered in this group. Target of the application of these processes are decreasing of remarkable assembly cost reduction and weight reduction by decreasing fastener and two sheet layered area for joining.

Monolithic structure or monolithic complicated part is also effective to long term durability. Free of joint means smooth load distribution and lower stress concentration level, that is good for fatigue life. And corrosion problem may be reduced because crevice corrosion and dissimilar metal corrosion etc. may be often caused around joint portion.

Our some recent studies about forming process are introduced in this report, these includes application of high form 6000 series alloy by cold working, precision casting, superplastic forming. These were examined the effectiveness of cost reduction and weight reduction by forming trial parts. Then mechanical properties and corrosion properties were examined.

2.NEW 6000 SERIES ALLOY OF HIGH FORMABILITY

2.1 Concept of the development of new alloy

A new high strength Al-Mg-Si-Cu alloy (6000 series alloy) was developed in Japan(Ref.1). Concept of this development is shown in Figure 1, this material is considered to replace the widely used 2024-T3.

Alloy 2024-T3 is one of Al-Cu-Mg alloys(2000 series alloy), and has been used as main aerospace structural material. The alloy has high strength and good resistance to fatigue crack growth. But it also has some problems in formability and the corrosion resistance. 6000 series alloys have lower strength compared with 2000 series alloys, but have some better properties like corrosion resistance, formability, lower density and lower production cost compared with those of 2000 series alloys. If the strength of 6000 series alloy is improved and the merits are maintained, the alloy may meet the requirements for the new major material. Alloy 6013 was developed with similar ideas, but tensile strength is not reached to that of 2024 –T3.

Mechanical properties and corrosion resistance were studied in some 6000 series alloys with varying amount of the alloying elements. Mainly Cu and Si were controlled, effect of them on strength is shown in Figure 2.

Finally chemical composition was fixed as shown in Table 1. This alloy is 3% lower density than 2024-T3.

2.2 Mechanical properties and corrosion properties of new alloy

Tensile and yield strength of new alloy is shown in Figure 3 compared with those of 2024-T3, 6013-T6 and 6061-T6. Tensile and yield strength of new alloy are larger than those of 6013-T6, and tensile strength is almost same as that of 2024-T3.

Figure 4 shows the comparison of fatigue crack growth rate between new alloy and 2024 –T3, these are almost same. And nearly the same fracture toughness data is obtained between new alloy and 2024 –T3.

Paper presented at the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

Corrosion properties of new alloy are compared with those of conventional alloys. Figure 5 shows the result of corrosion test per ASTM G-110. 2024-T3 shows very severe corrosion attack, and 6000 series alloy shows shallow intergranular corrosion. Corrosion depth of new alloy is smaller than that of 6013-T6.

Above these tests and another tests, for example, bearing test and shear test etc. show that new material is promising mechanical properties used as replacing 2024 alloy. This new alloy may have more long term durability than 2024 alloy because of better corrosion properties, and almost same fatigue crack growth and toughness.

2.3 Formability test and trial part forming results etc.

New material is formed at T4 condition and then aged to T6 condition. Formability of new alloy is compared with that of conventional alloys, hemispherical dome test and bending test were carried out. Result of hemispherical dome test is shown in Figure 6. New alloy shows larger forming height than that of 2024-T3 or 2024-W, and almost same as that of 6013-T4. And bending test shows same trend, permissible bending radius of new alloy is smaller than that of 2024-T3 or –W, and same as that of 6013-T4.

Effect of the strain of cold forming on mechanical properties of 2024-T3 is very large, but that of new alloy is small because aging process moderates cold working effect. Effect of strain on compressive yield strength is shown in Figure 7 as example.

So, this material may be useful to severe contoured panel, bead panel and curved frame etc.. Figure 8 shows bead panel of new 6000 series alloy sheet. Inspection is secured no irregular thickness nor distortion. In this case, about 50% cost reduction may be estimated by changing the design concept from assembled sheets to monolithic bead panel.

Precision extrusions and hollow extrusions of new alloy are also developed, these are shown in Figure 9. Dimension of wall thickness is able to be settled very thin. So these may be used effectively as light and high stiff components.

Another merits of this new alloy are weldable and good thermal stability of strength. For the latter property, tensile strength is a little lower than that of 2219-T8, but thermal stability of strength is rather better, because aging temperature is high. So this alloy may be applied to welded structures or high temperature structures in aluminum used field.

3. LARGE PRECISION CASTING OF ALUMINUM ALLOY

3.1 Development of casting alloy and casting process

It is spreading to apply premium-quality large size precision casting for primary aircraft structures. D357,A357 and A201, which are used as casting alloys, have good mechanical properties.

Improved casting methods such as low pressure casting enables to decrease casting discontinuities, these are casting cavities and cracks, and realizes the complicated thin casting products. And control of microstructure is important to get good strength, Figure 10 shows the effect of secondary dendrite arm spacing on strength of A357 cast alloy. Strength decreases with increase of arm spacing. Arm spacing decreases with increase of cooling rate in solidification, so chill block in a mold should be used if high strength may be secured partially in the structure.

Following heat treatment process after casting process is also important, uniform cooling rate must be requested to get distortion free and proper strength cast structure.

3.2 Mechanical properties and corrosion properties of cast alloys

Figure 11 shows the yield strength and fracture toughness of cast aluminum alloys compared with sheet or forged Al alloys. Fracture toughness of A357 is not so high compared with that of 2024 sheet, but yield strength of A357 is almost same as that of 2024-T3 sheet, and fracture toughness and yield strength of A201 alloy is almost same as that of 7075 forging. Strength of D357 is not shown, it is same as that of A357, and rather narrower strength dispersion. A357 may used to replace assembled sheet parts of 2024-T3 sheet, however applied structure must be the high stiff structure of low stress level and smooth stress distribution. A201 may be used to replace mainly 7000 series structural parts. In this report, results of A357 are introduced as follows(Ref.2).

Figure 12 shows the fatigue crack growth rate of A357 cast alloy, which inner discontinuities are permissible level examined by X ray. Crack growth rate is relatively lower than that of 2024-T3 or 7075-T6 wrought material. And Figure 13 shows the fatigue strength of A357 alloy. Fatigue strength is lower than that of 2024-T3. When single crack is propagated in the case of fatigue crack growth test, small casting cavities may trap crack tip. So crack propagation path becomes wavy, then it makes fatigue growth slow. On the contrary, many cracks are caused at casting cavities simultaneously in the case of fatigue strength test , and these cracks are joined each other. This phenomena decreases fatigue strength. When strength level is effectively low, former fracture mode of slow crack propagation from single or a few portion may be considered as structural fracture mode.

General corrosion property of A357 is said to be better than that of 2024-T3 bare sheet. And A357 does not show susceptibility of stress corrosion cracking.

Long term durability of A357 or D357 cast structure may be considered that it depends on design philosophy to fit fatigue properties. Casting structure should be designed by stiffness driven and stress in the structure should be low level.

3.3 Study of application

Application of precision casting proves advantage of the weight reduction and cost saving available when moving from complicated fabricated structures to monolithic integrated structures.

Figure 14 shows the exit door manufactured by casting for trial, size is about $600 \times 620 \times 70$ mm and thinnest portion is $1.5mm^t$. Material is A357. And then Figure 15 shows comparison of cost and weight between precision casting and assembled structure. Thickness of casting structure must be increased compared with sheet thickness of assembled structure because of casting factor. However 10% weight reduction may be attained because there is no fastener nor two

sheets layered area for joining.
Cost may be decreased to about a half of the cost of assembled structure of sheet.

Precision casting is promising method, but designers must consider several problems of effective application when they try to apply it. These problems are special design concepts which include the knowledge of liquid metal flow and cooling characteristics in mold, weight saving design technique, and then knowledge of quality assurance concepts to casting defects. More strict contact among staff from design to qualification must be required than usual process because effect of mutual relationship among design and each process on the quality of products is stronger than usual process.

Now, D357 becomes available, and application study of casting is developed widely, and structures about 2mϕ is studied as casting application.

4. SUPERPLASTIC FORMING OF ALUMINUM ALLOY

4.1 Outline of superplastic forming method

Superplastic forming technology makes it possible to fabricate a sheet of complex configuration such as deep bulge forming, and this technology also enables to change assembled sheet parts to an integrated sheet part. Fine-grained aluminum alloys such as 7475 and 5083 alloy show superplasticity at restricted narrow range of both temperature and strain rate. Figure 16 shows the relation between testing temperature of tensile test and elongation of fine-grained 7475 alloy, which shows superplastic phenomenon at about 500℃.

Forming method used to application is generally bulge forming by air pressure, sheet to be formed is fixed in jig and sequentially controlled gas pressure to get desired strain rate is loaded at proper temperature. Smooth plastic metal sheet deformation on die must be required, so lubricating media between sheet and die is also important.

4.2 Mechanical properties of superplastically formed part

Mechanical properties are same if defects are not caused during forming. When forming condition is improper, forming is not completed or thickness distribution becomes improper.

Another problem is the generation of micro cavities as inner defects which decrease strength. Figure 17 shows the effect of the amount of cavities on tensile properties of formed 7475 alloy. When cavity area is over 1% of the examined unit area of cut section, tensile strength and yield strength and then elongation become decreased and lower than the values of specification. And Figure 18 shows the effect of the amount of cavities on fatigue properties. Fatigue strength become decreased when cavity area is also over 1% (Ref.2).

Amount of cavities depends on prepared materials and process. Microstructure etc. is secured by material specification. Amount of cavities increases with increase of elongation so, it is required not to cause partially heavy elongation by means of design modification and proper process control. Corrosion properties are not affected by the generation of cavities.

From the above, quality control about thickness distribution and amount of cavities should be important to secure long term durability of superplastically formed part and these factors depend on process. Dummy part to be adjusted severest elongation level of real part may be formed simultaneously to check the generation of cavities as sampling inspection.

4.3 Application study of superplastic forming

Figure 19 shows the trial product of superplastic forming, which material is 7475 alloy. Thickness distribution is proper and cavities are very few. Most important design requirement is structural stiffness alike casting, and it is confirmed. Many parts are integrated to one part, so about 20% production cost reduction is predicted in this case.
When designers would like to use this method, they must consider special design technique about thickness control and quality assurance method about generation of cavities etc. and strict contact with production stuff as is the case of casting.

5. CONCLUSIONS

Application of three new processes of aluminum alloys to pursue affordability are studied, these are application of high strength 6000 series alloy of high formability, large precision casting and superplastic forming. Concluding remarks are as follows.

(1)Removing joint and assembly process by using these new processes to form monolithic structure is very effective to attain total cost reduction and weight reduction.

(2)Monolithic structures by these new processes have basic merits of long term durability, that is, free of joint which cause many problems. These are crevice corrosion and dissimilar metal corrosion, then bearing fatigue strength and not smooth stress flow.

(3)New 6000 series alloy developed in Japan has good mechanical properties to replace 2024-T3 alloy. Fatigue strength and corrosion properties are also good.

(4)Structural stiffness is often most important design requirement for the application of precision casting and superplastic forming. Fatigue strength properties of structure applied these two methods are good if design is fair and inner discontinuities are proper low level.

(5)These new forming methods often require new design concept and new qualification method etc. So, more strict contact among staff of design, material, manufacturing and qualification is required.

6. REFERENCE

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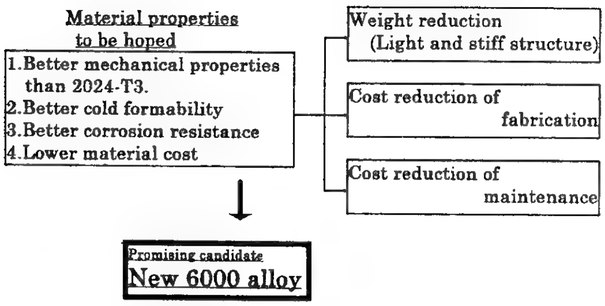


Figure 1 Concept of the Development of New 6000 Series Alloy

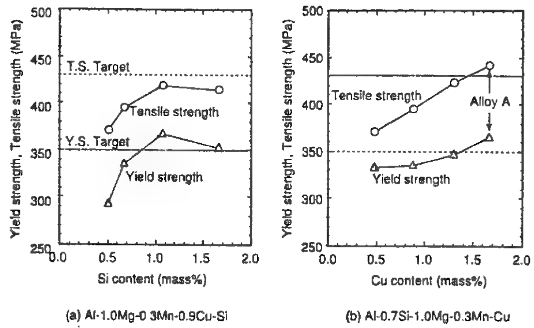


Figure 2 Effect of Silicon and Copper Content on Strength of Al-Mg-Si-Cu Alloys

Table 1 Chemical Composition of New 6000 Series Alloy (mass%)

Si	Fe	Cu	Mn	Mg	Cr	Zn	Ti	Al
0.75	0.14	1.64	0.01	1.01	0.15	0.01	0.02	bal.

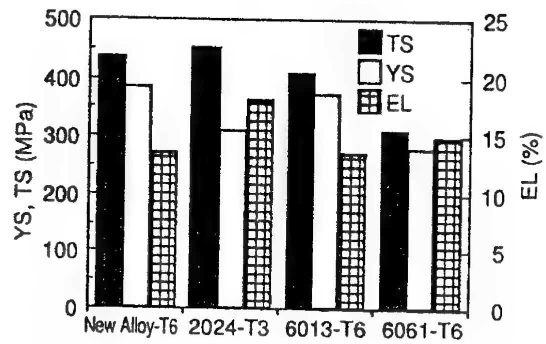


Figure 3 Mechanical Properties of New Alloy and Conventional Alloys (Thickness:1.27mm)

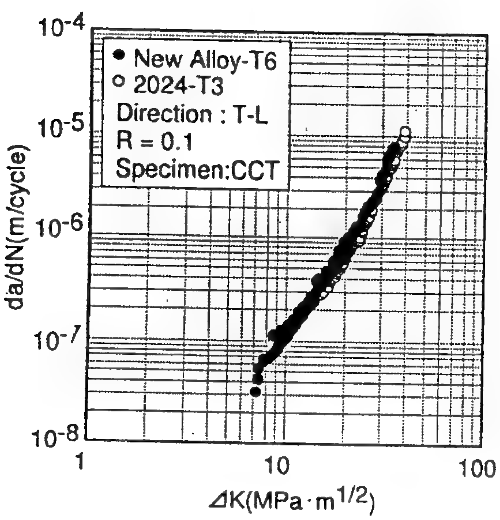


Figure 4 Comparison of Fatigue Crack Growth Rate between New Alloy and 2024-T3 (Thickness:1.27mm)

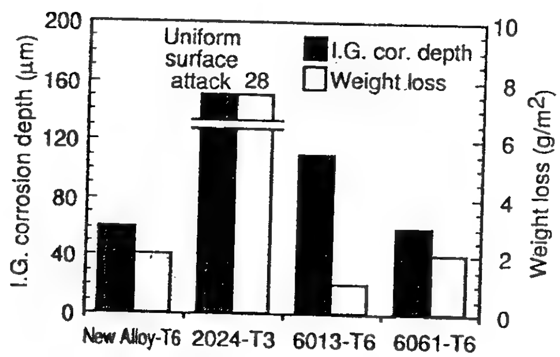


Figure 5 Intergranular Corrosion Depth and Weight Loss of New Alloy and Conventional Alloys (ASTM-G-110)

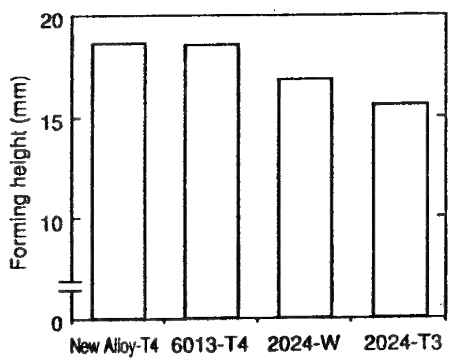


Figure 6 Comparison of Hemispherical Dome Test of New Alloy and Conventional alloys (50mm Diameter)

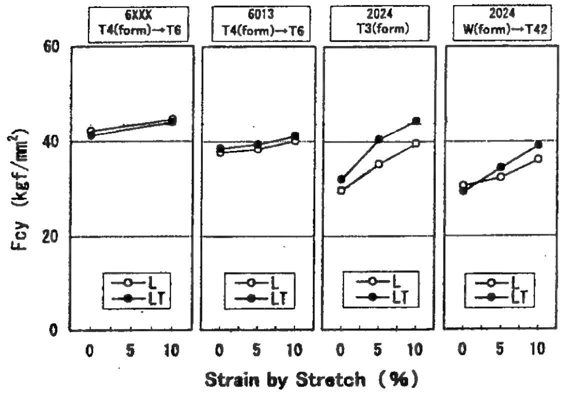


Figure 7 Effect of Strain by Stretching on Compressive Yield Strength of New Alloy and 2024 Alloy

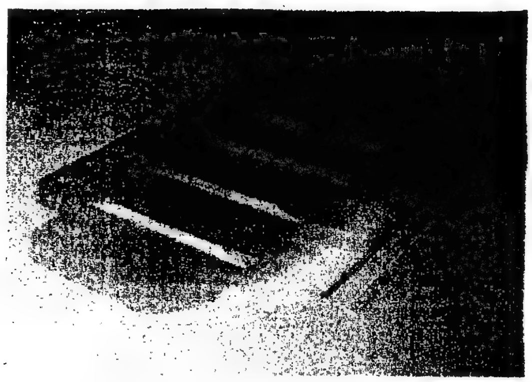


Figure 8 Bead Panel Trial Part of New Alloy (Bead Height:8mm,Width:27mm)

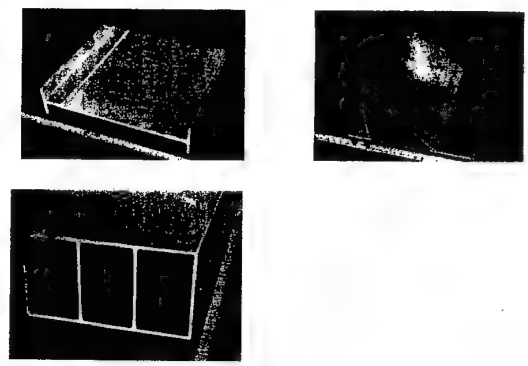


Figure 9 Precision Extrusion and Hollow Extrusion of New Alloy

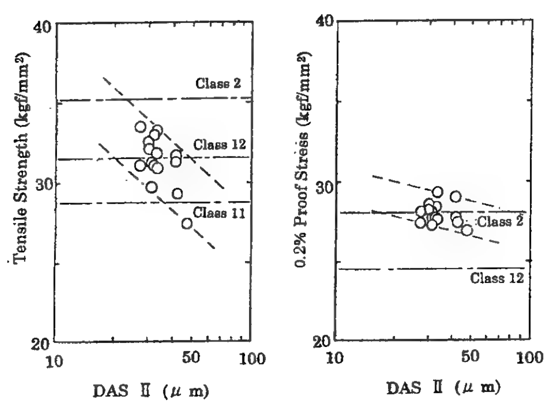


Figure 10 Effect of the Secondary Dendrite Arm Spacing (DAS Ⅱ) on Tensile Properties of A357 Cast Alloy (Strength Class: from MIL-A-21180)

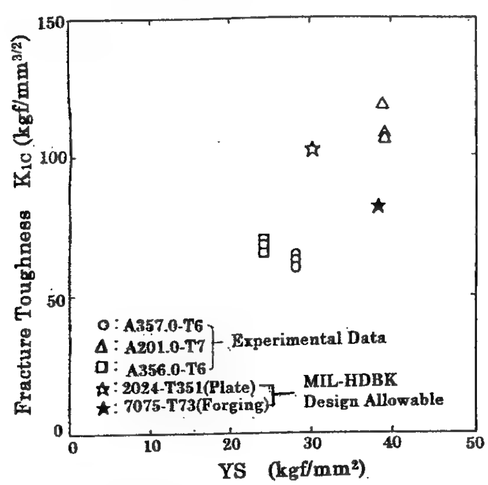


Figure 11 Relationship between Tensile Yield Strength and Fracture Toughness of Aluminum Cast Alloys and Conventional Wrought Alloys

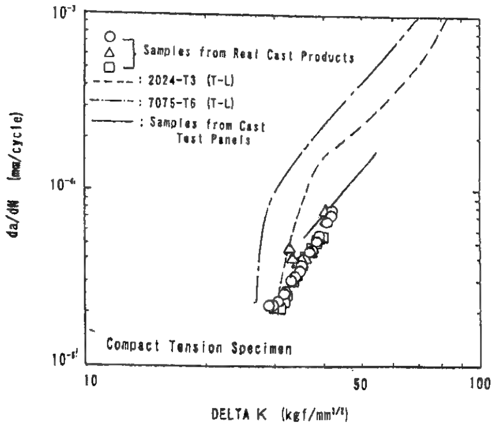


Figure 12 Fatigue Crack Growth Rate of A357 Cast Alloy Compared with That of Wrought Alloys

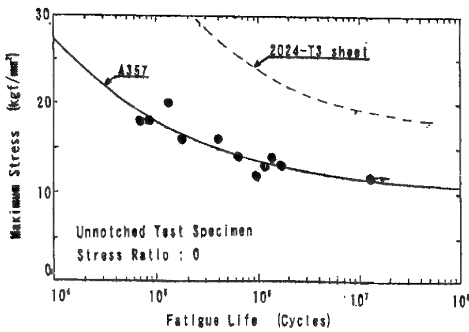


Figure 13 Fatigue Strength of A357 Cast Alloy Compared with That of 2024-T3 Sheet

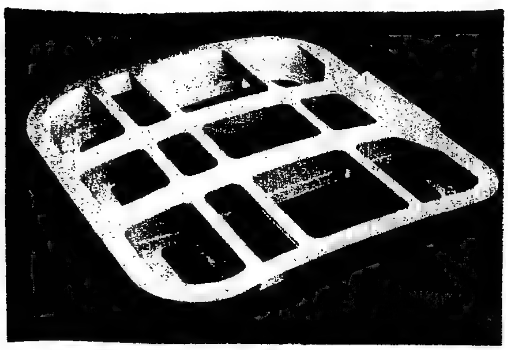
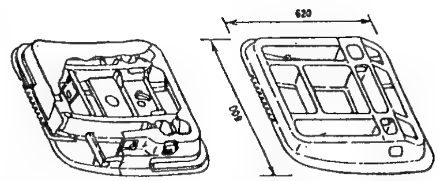


Figure 14 Exit Door Manufactured by Casting for Trial (Material: A357)



	Estimated weight (%)	Estimated cost (%)
Assembled structure of sheet	100	100
Precision casting	91	53

Figure 15 Comparison of Predicted Cost and Weight between Precision Casting and Assembled Structure

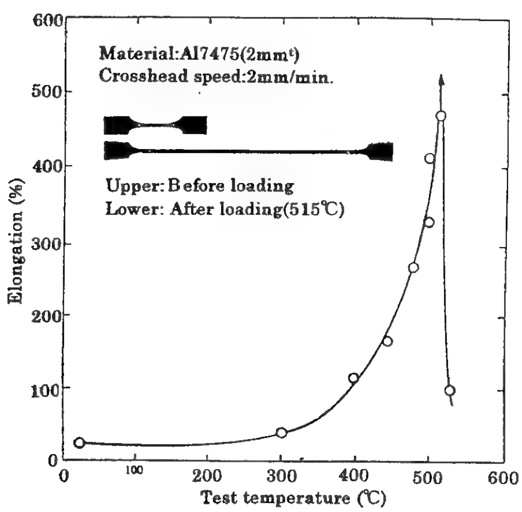


Figure 16 Superplastic Phenomenon of Fine- grained 7475 Alloy

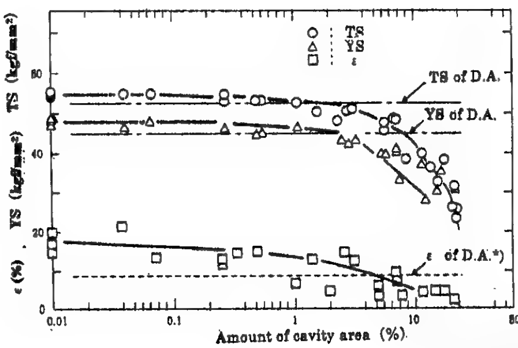


Figure 17 Effect of the Amount of Cavities on Tensile Properties of Superplastically Formed 7475 Alloy (Heat Treated to T6 after Forming)

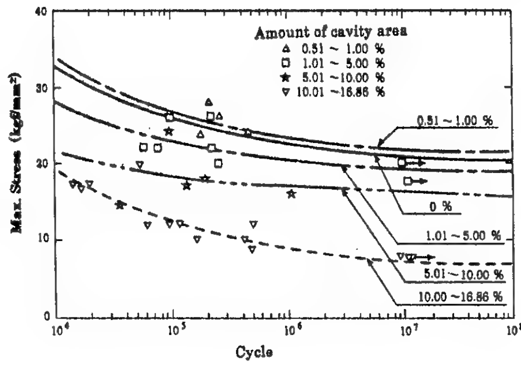


Figure 18 Effect of the Amount of Cavities on Fatigue Strength of Superplastically Formed 7475 Alloy (Heat Treated to T6 after Forming)

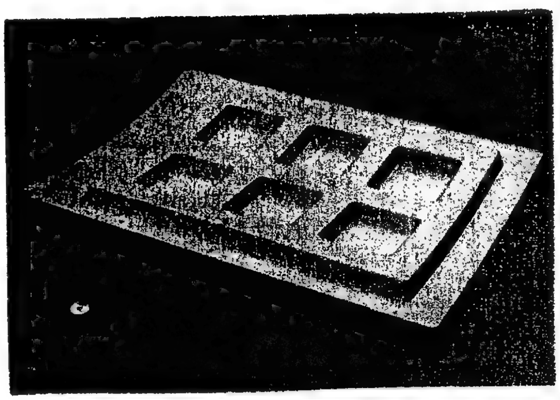


Figure 19 Trial Product of Superplastic Forming

Plastic envelope in propagating crack wake on Al-Li alloys subjected to fatigue cycles and to different heat treatments

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Summary. The aim of this research is to study the fracture behaviour of three Al-Li alloys (2091-2195-8090), using standard CT specimens, in the frequency range of 1-10 Hz. Each of these three alloys is subjected to different heat treatments and its homogeneity is analysed, before and after treatments, by Scanning Electron Microscope (SEM). Crack tip opening displacements and plastic zone envelope analyses are fully treated by experimental and numerical results and fatigue crack growth process is extensively reported. At the end of fatigue tests, specimen fracture surfaces have been deeply analysed by SEM in order to individualise the characteristics of fracture as function of frequency, ΔK and load ratio R.

Key words: crack growth, plasticity, CT specimen, SEM.

1. INTRODUCTION

Renewed and extensive research and development activities led to new generation of Al-Li alloys by three major producers: Alcon, Alcoa and Pechiney. For these alloys the improvements in various properties, including density and stiffness, result from lithium additions and have the potential to save up to 10% in weight by direct substitution, and up to 18% in weight if the increased specific stiffness (modulus/density) is performed [1] [2].

Commercial aluminium-lithium alloys are targeted as advanced materials for aerospace technology primarily because of their low density, high specific modulus, and excellent fatigue and cryogenic toughness properties. The superior fatigue crack propagation resistance of Al-Li alloys, in comparison with other traditional alloys, is primarily due to high levels of crack tip shielding, meandering crack paths and resultant roughness-induced crack closure.

However, the fact that these alloys derive their superior properties extrinsically from the above mechanisms has certain implications with respect to small crack and variable amplitude behaviour. For example, aluminium-lithium alloys loose their fatigue advantage over conventional aluminium alloys in compression dominated variable amplitude fatigue spectra tests. However, in tension dominated spectra, aluminium-lithium alloys show greater retardation on the application of single peak tensile overloads. The principal disadvantages of peak strength aluminium-lithium alloys are reduced ductility and fracture toughness in the short transverse direction, anisotropy of in plane properties and accelerated fatigue crack extension rates when cracks are microstructurally small. These limitations have imposed the direct substitution of aluminium airframe alloys with aluminium-lithium alloys, although it is possible to group the present aluminium alloys and the current aluminium-lithium alloys in terms of product form and of primary design criteria.

During the past 20 to 30 years influences on crack growth behaviour have been systematically investigated. It was found that the mean stress applied is important and its effect is closely correlated to crack closure behaviour. In recent years the threshold behaviour had also been extensively investigated. Doker [3] showed that most of the da/dN vs. ΔK data for different materials which are presently available fall below some limiting curves. The experimental data show that they fall comparatively close together for different R ratios in the middle part of the curve (mean stress). At the lower and higher ends of the curve, data for different material considerably deviate according respectively to crack closure effect and to fracture toughness K_c values.

The aim of this research is to investigate, by means of experimental and numerical analyses, fatigue behaviour of 2091, 8090 and 2195 aluminium-lithium alloys and it has mainly two purposes: 1) to define frequency effects on crack propagation rate and 2) to survey plastic zone

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progress, as function of time and space, over crack tip. First purpose was carried out by experimental tests. These were all performed in the same operating conditions: that is, same environment (lab air), same loads, same specimens geometry whereas the only variable parameter was the frequency. Testing frequency range has been 1-10 Hz.

The second part required use of an elasto-plastic analytical model, which has actual crack length (experimental data) as input and gives out, as output, the plastic zone size together with length, width and abscissa of all elements (totally ten) used to describe it. A comparison will be carried out 1) among different alloys, for the same testing conditions, and 2) for the same alloy, for different frequency values. The research is concluded with presentation of fracture surface analysis by SEM.

2. THE PROPOSED APPROACH

A two dimensional, weight function based, non linear elasto-plastic analytical model has been used for considering plastic zone evolution. This model is restricted to two dimensional cracked bodies with cracks under mode I loading. Using predictions of crack opening stress, the effective stress intensity factor range, at fatigue crack tip under a given cyclic loading, is computed. The effect of residual stress on fatigue crack propagation is of great practical significance and has been the focus of much research. Superposition techniques are often used when assessing the effects of fatigue crack propagation. The superposition involves the computation of the stress intensity factor, which is associated with the initial pre-existing *residual* stress field. This factor is then superposed upon the stress intensity factor that results from *external* loading, to give the total stress intensity factor as:

$$K = K_{res} + K_{ext} \tag{1}$$

Maximum and minimum values of the total resultant stress intensity factor K are computed for the cyclic loading and a total resultant stress intensity factor range ΔK is calculated. This resultant stress intensity factor range may then be used to compute the predicted fatigue crack growth rate da/dN using a $da/dN=f(\Delta K)$ correlation.

The superposition technique is used extensively because of its simplicity. It has been criticised by some researchers because it considers only the initial residual stress field that exists in the uncracked structure, with no acknowledgement of the redistribution of residual stress that occurs as the propagating fatigue crack penetrates, with its free or partially free surfaces, the residual stress field. Other researchers have argued that the redistribution of residual stress is of no consequence.

Bueckner (1958) has demonstrated mathematically that, for linear elastic materials, stress intensity factors resulting from a given applied loading may be computed using the stress distribution in the uncracked structure. Heaton [4] has presented a mathematically rigorous proof that generalises Bueckner's formulation to include both thermal and residual stress fields. These works suggest that, for linear elastic materials, the redistribution of applied and residual stresses due to fatigue crack propagation is of no consequence when computing stress intensity factors and fatigue crack growth through use of correlation $da/dN=f(\Delta K)$.

These conclusions are applicable to linear elastic materials only. The existence of plastic deformation at the crack tip, even under small scale yielding conditions, will produce crack generated residual stresses at the crack tip and crack closure along the crack surfaces of a propagating crack. The superposition methodology is unable to account for the influence of these non linear plasticity effects. Consequently, the use of linear elastic superposition techniques for prediction of the effects of residual stress on fatigue crack propagation and plastic zone evolution, may result in predictions that correlate poorly with experimental observations. More accurate predictions may be possible if an increased understanding is established of the interactions between residual stress fields and crack closure.

The model used here is an extension of Newman's Model [5]. While Newman investigated specific two dimensional geometry, the model described here accommodates the analysis of any two dimensional geometry that exhibits a crack emanating from a free surface for which a weight function is known.

Let's consider a propagating fatigue crack being closed by plastic deformation at crack tip and propped open by a wake of plastically deformed material along the crack surfaces. A series of rigid perfectly plastic elements is used to model the residual plastic deformation along the fatigue

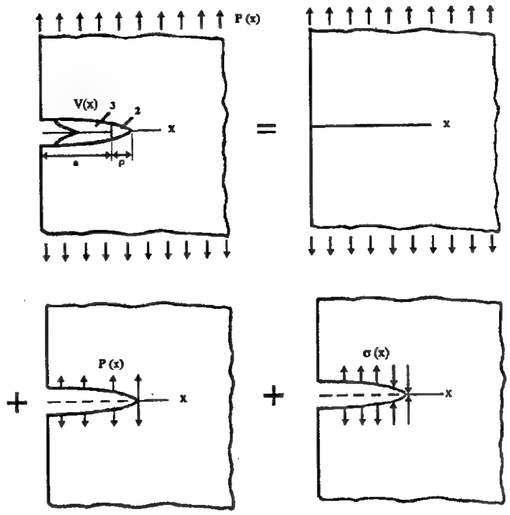

Figure 1: Elasto-plastic fatigue crack model as superposition of elastic ones

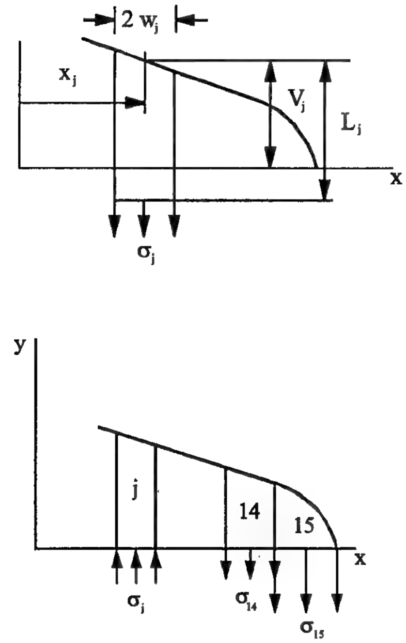

Figure 2: Elements and applied stresses

crack surfaces and the plastic zone ahead of the propagating crack. The model is constructed using a superposition of two elastic crack problems and is based on the Dugdale Model, modified to leave plastically deformed material along the crack surfaces as the crack advances.
The external loading is assumed to vary between a fixed maximum value S_{max} and a fixed minimum S_{min}; thus only constant amplitude applied loading is considered. Stress $\sigma(x)$ is generated along crack surfaces as a consequence of plastic deformation at the crack tip and of residual plastic deformation along the crack surfaces. The model is composed of three region as indicated in Fig. 1.

1. A linear elastic region containing a fictitious crack of length $d = a + \rho$;
2. A plastic region of length ρ ahead of the actual crack
3. A region of residual plastic deformation along the crack surfaces

The plastic and residual plastic deformation regions have been modelled using n rigid perfectly plastic bar elements as shown in Fig. 2. To approximate the effects of strain hardening, the flow stress σ_0 is taken to be the average value between the yield and the ultimate strength. The generic stress σ_j is applied on the linear elastic region and is transmitted to this region by the n bar elements as illustrated in Fig. 2.

As originally postulated by Dugdale, under a tensile stress $\sigma(x)$, element 15 would yield in tension and crack opening would be restrained.
Generic element j is located along and attached to the upper crack surface. Under low levels of $\sigma(x)$, crack remain closed at j, placing element j in compression, and thus propping open the crack as shown. Under higher levels of $\sigma(x)$ the crack would open and $\sigma_j \rightarrow 0$. The plastic zone ahead of the current crack tip under the maximum applied loading is modelled using ten elements. The plastic zone size is computed assuming finite stresses at the fictitious crack tip as originally postulated by Dugdale. Element widths w are assumed such that elements near the actual crack tip are smaller in width, while element lengths L, in the plastic zone, are given by weight function based computation. An initial edge crack length a is assumed to exist and it is modelled using elements 1-5, with element 1 located at the crack mouth. These elements have negligible length.
Weight function $m(x, a)$, which remains valid for $a/W < 0.5$ [6], and crack tip opening displacement $V(x_i)$ expressions are as follows:

$$m(x, a) = \frac{1}{\sqrt{2\pi(a-x)}}\left[1 + m_1\left(1 - \frac{x}{a}\right) + \right.$$

$$+ m_2\left(1-\frac{x}{a}\right)^2\Bigg]$$

$$V(x_i) = \frac{2}{E'}\int_{x_i}^{d} m(x_i,\alpha)K_I(\alpha)d\alpha \qquad (2)$$

where m_1 and m_2 depend upon the ratio a/W between specimen width W and crack length a, α is a fictitious crack length, E' is the generalised Young modulus and K_I, which is the mode I stress intensity factor, is given by:

$$K_I(\alpha) = 2\int_0^{\alpha} T(x)m(x,\alpha)dx \qquad (3)$$

Combining and substituting the previous equations, the total generic displacement V_i results as

$$V_i = f(x_i, S_a) - \sum_{j=1}^{n}\sigma_j g(x_i, x_j) \qquad (4)$$

where the first term is the displacement due to external loading S when crack lenght is a and the second term represents the displacement in x_i due to the stress σ_j on the generic j element. The crack extension creates a new element with width w_i equal to Δa and length L_i equal to the crack tip opening displacement V_i before extension. In fact under maximum loading crack surfaces are completely opened and element lengths L_i result equal to surface displacements V_i. Under the minimum applied stress, elements representing the residual plastic deformation along the crack surfaces may come into contact and transmit a compressive stress; compressive yielding is also possible, requiring the computation of new element lengths. The stress along the line of crack propagation, that produces the first full opening of the crack, is defined as opening stress $\sigma_0 h(x)$, where σ_0 is a scaling parameter that defines the magnitude of the crack opening stress and $h(x)$ is a shape function which defines the crack opening stress along $y = 0$ (line of crack propagation). Under a uniform stress with no stress concentration, one obtains $h(x)$=1, while it varies if the geometry of interest exhibits a stress concentration.

Under constant amplitude loading, the last element to loose contact and open is generally the element along the crack surface closest to the crack tip. It is assumed that the only portion of the loading cycle that contributes to the crack propagation is the portion for which the crack is fully open, as originally postulated by Elber [7]. Small fatigue cracks induce limited plastic zones mostly providing plane strain conditions. As crack propagates and plastic zone size increases, a transition to plane stress condition can be observed. It results that crack growth analysis requires the modelling of both plane strain and plane stress conditions. In light of the introduction of a correction factor, which varies between 1 and 3, as demonstrated by Newman [5], the present model predicts accurately the crack propagation.

3. THE EXPERIMENTAL TESTS

The elasto-plastic model is used to survey plastic zone progress, as function of time and space, in a cyclically loaded edge cracked panel (CT specimen) as shown in Fig. 3. The constant amplitude loading was a uniform pulsating tension with R=0.3 and S_{max}=400 Kg. The panel width W is 40 mm. and an initial crack size a=6 mm. is assumed. The specimens used in the experimentation have been obtained in the LT configuration. The preparation of the surface has been particularly studied to facilitate the optical reading of the crack length; Knuth-Rotor System, that consists in using some rotating disks at constant speed on which are supported abrasive SiC papers, has been utilised. The cooling has been performed through a continuous throw of water, while the polish by some rotating cloths wetted by suspensions of allumina particles of 1 μm diameter. Finally, on both specimen faces some imprints, spaced 1 mm the one from the other, have been realised. The ultimate strength for the three alloys, in LT direction, are: 462 MPa for 2091 alloy, 557 MPa for 2195 alloy and 470 MPa for 8090 alloy.

Their section is reported in succession in Fig.4. Like other age-hardened aluminium alloys, aluminium-lithium alloys achieve precipitation strengthening by thermal ageing after a solution heat treatment. The age hardening of aluminium-lithium alloys involves the continous precipitation of δ' (Al_3Li) from a supersaturated solid solution. Aluminium-lithium based alloys are microstructurally unique. They differ from most of the aluminium alloys in that once the major strengthening precipitate (δ') is homogeneously precipitated, it remains coherent even after extensive ageing. All three alloys under examination have a temper designation of T8,

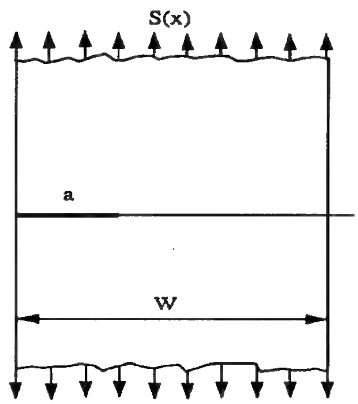

Figure 3: Finite width panel with edge crack a

which consists in a solution heat treatment, a cold working and an artificially ageing. Two of them, 8090 and 2091 alloy, have been subjected to a further treatment, designated as Tx51 (normally applied to plate and to rolled or cold finished rod and bar), which relieves residual stresses by stretching.

The measures of crack length have been performed by an optical microscope WILD M38, provided of an enlargement of 40X. Measures have been taken on both specimen sides and, periodically, each specimen has been removed and observed with the electron microscope or with an optical microscope that allows bigger enlargements.

Besides the necessary data for the determination of wake plastic strain, i.e. length, width and abscissa (as function of cycles number) of each of the ten elements, a parameter Ψ, which takes into consideration the crack opening and consequently keeps track of ΔK_{eff} and not simply of ΔK, is considered as,

$$\Psi = \frac{\Delta K_{eff}}{1 - S_0/S_{max}} \tag{5}$$

Finally, the fracture surfaces have been observed by SEM (Scanning Electron Microscope). The same surfaces, attached subsequently by a specific acid solution (50 ml H_2O, 50 ml HNO_3, 32 ml HCl, 2 ml HF), have been studied with the scanning microscope for the individualisation of fracture plans and of flow directions.

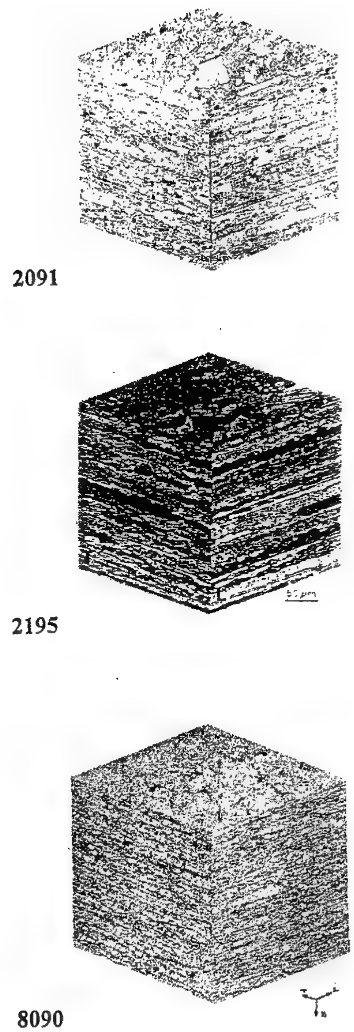

Figure 4: Section of the alloys under examination

4. RESULTS

The tests performed for the 8090 alloy are reported in Figg. 5-6. These allow to follow the crack evolution both in terms 1) of crack length vs the number of effected cycles and 2) of the evolution of plastic zone adimensional size ρ vs the number of cycles. From Fig. 5 is deduced, as confirmed by most bibliography, that, as frequency increases, the rate of propagation decreases and the ultimate number of cycles increases. Such conclusions are also confirmed by Fig. 6, where as frequency increases, the dimension of the plastic zone ρ (and therefore the degree of local plasticity present at the crack tip) tends to decrease.
The tests performed for the 2195 alloy are reported in Figg. 7-8: same conclusions are valid both in terms of crack rate of propagation and in terms

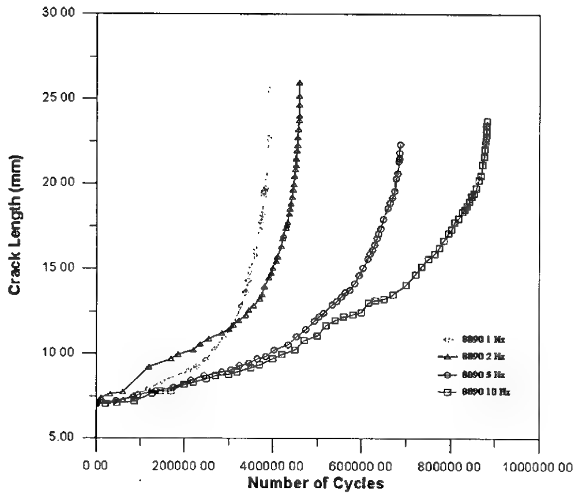

Figure 5: 8090 alloy

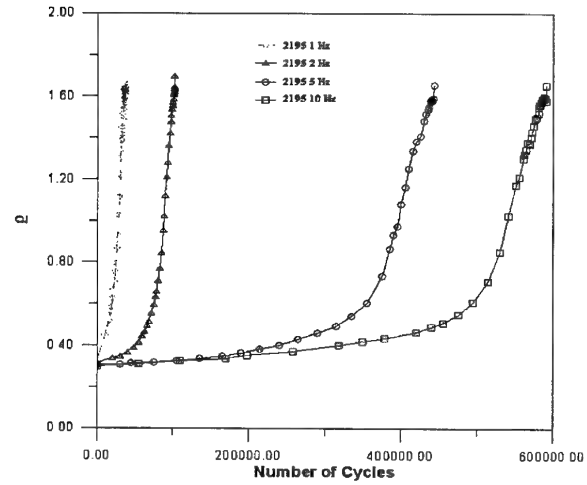

Figure 8: 2195 alloy

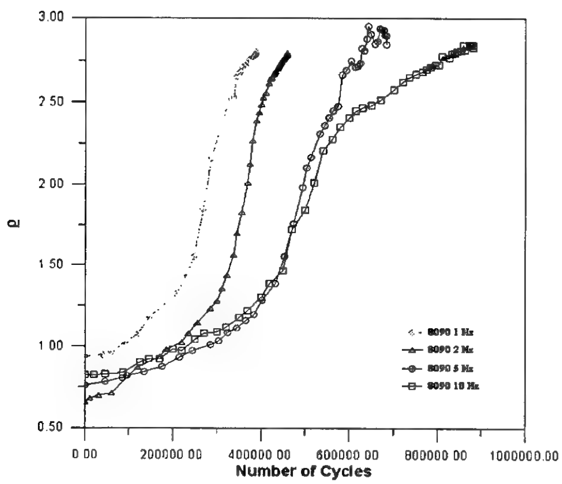

Figure 6: 8090 alloy

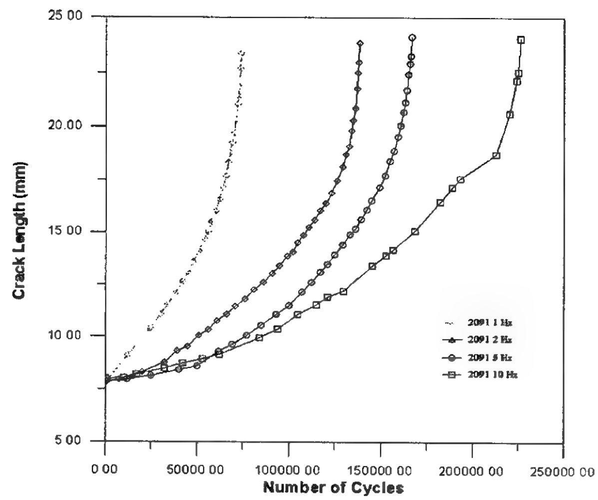

Figure 9: 2091 alloy

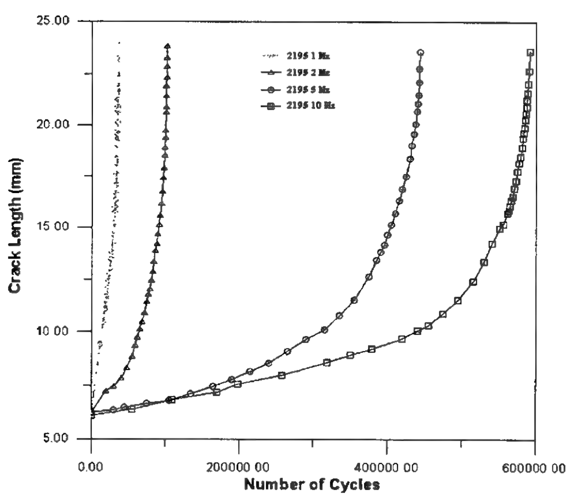

Figure 7: 2195 alloy

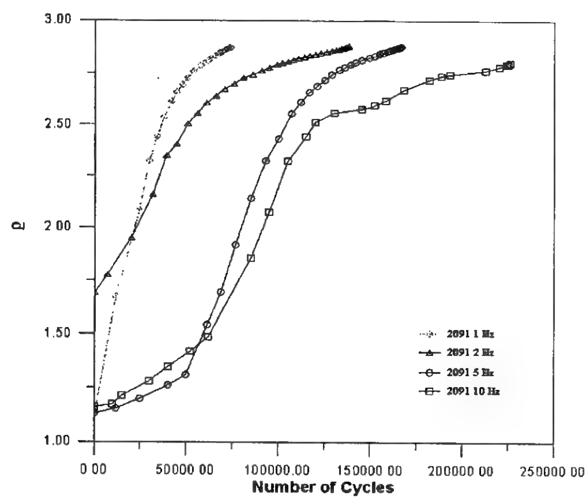

Figure 10: 2091 alloy

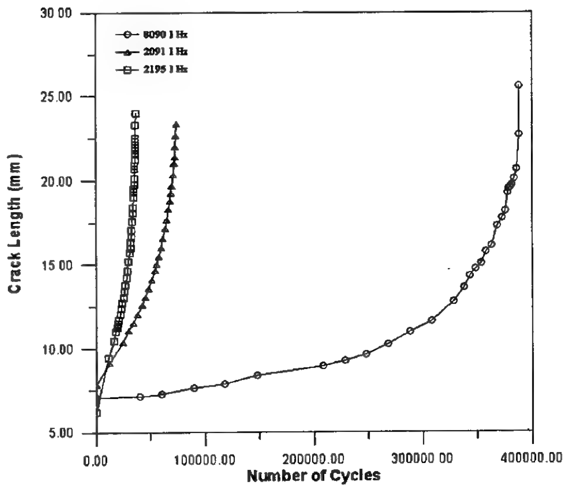

Figure 11: Comparison at 1 *Hz*

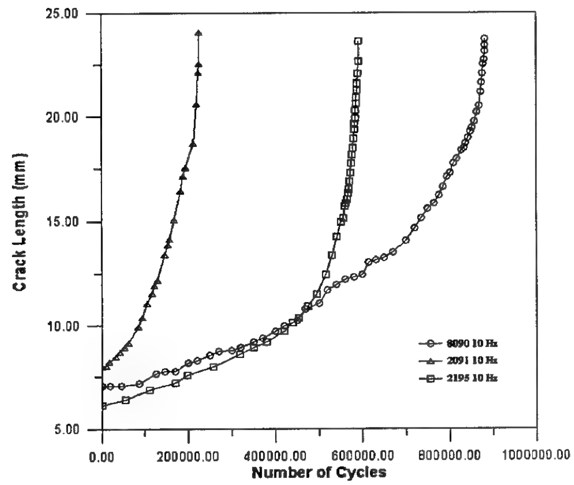

Figure 14: Comparison at 10 *Hz*

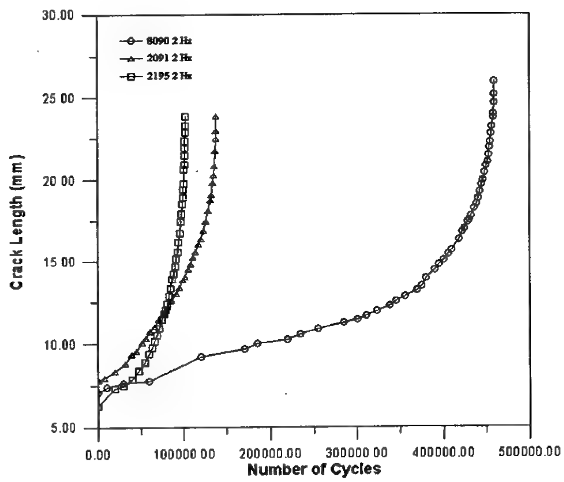

Figure 12: Comparison at 2 *Hz*

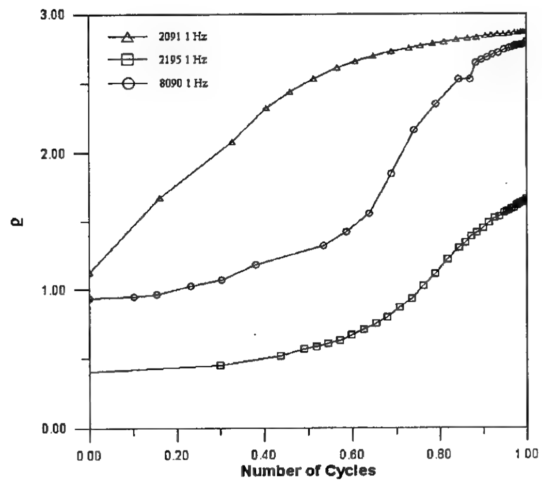

Figure 15: Comparison at 1 *Hz*

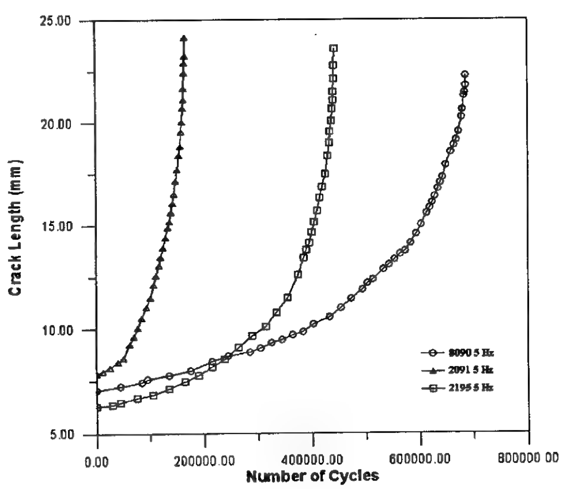

Figure 13: Comparison at 5 *Hz*

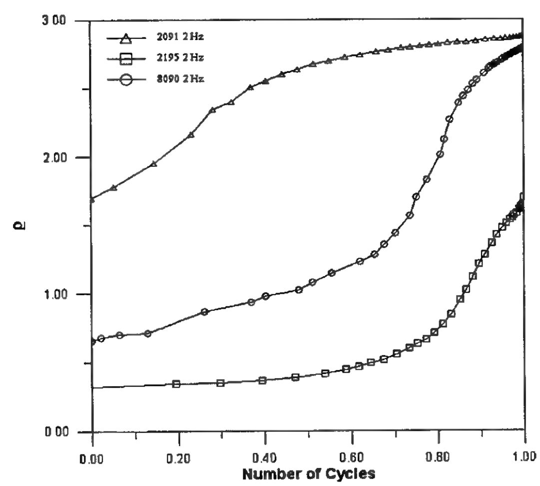

Figure 16: Comparison at 2 *Hz*

of plastic zone size. For such alloy, however, the gap between the curves at low frequency (1-2 Hz) and the curves at higher frequency (5-10 Hz) results comparably high, denoting a remarkable frequency effect even if a larger ammount of tests at lower and higher frequencies should still be provided. Finally the tests performed for the 2091 alloy are reported in Figg. 9-10. Similar results are obtained even if this alloy introduces more regular variation between frequencies.
In Figg. 11-12-13-14 are represented some comparative diagrams between the considered alloys. In order to facilitate a correct comparison, the curves are reported with frequency as parameter. Fatigue life for 8090 alloy is greater at any frequency, resulting much more evident at lower frequency.
In Figg. 15-16-17-18 plastic zone size (ρ) is plotted as function of the adimensional cycles number (n/N). In this case 2195 alloy, which shows better mechanical performances (in terms of yield and ultimate stress), seems to be less sensitive to the phenomenon of plasticity, always present at crack tip. In this context, 2091 alloy shows an extreme sensibility to plasticity, denoting, expecially at higher frequency, the largest plastic zone size. 8090 alloy assumes an intermediary role between the previous two (apart of the test at 10 Hz for n/N>0.8) showing a rather regular trend.
Wake plastic zone progress, which represents residual stress distribution on crack surfaces, is analysed through the monitoring of the first element length vs crack propagation. Generally its exponential behaviour has been found to change passing from 1 to 10 Hz, meaning that wake plastic zone is not only spreading on crack surfaces but also varying its shape. For the 2195 alloy, on the contrary, the exponential fitting curve (Fig. 19) remains a Vapor Pressure one, having a standard error close to 1 10^{-5} and a correlation coefficient of 0.999.

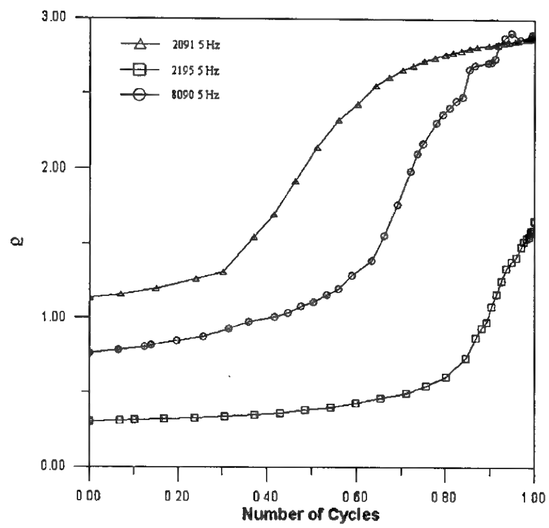

Figure 17: Comparison at 5 Hz

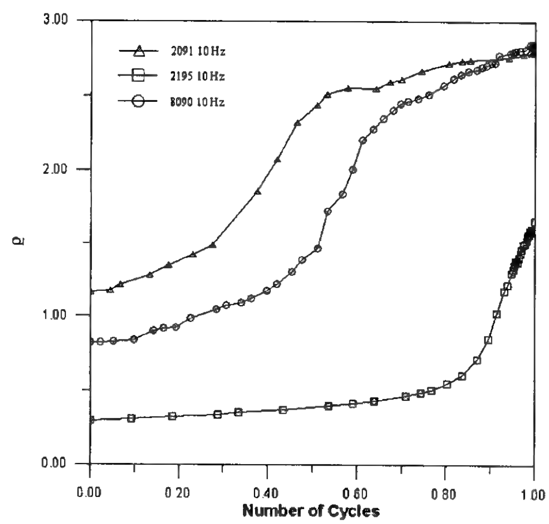

Figure 18: Comparison at 10 Hz

5. CRACK SURFACE ANALYSES

At the end of fatigue tests, specimen fracture surfaces have been analysed by SEM in order to individualise the characteristics of fracture as function of the frequency and ΔK.
The fatigue fracture of 8090 alloy could be investigated through two different models: Shear Facet Mode and Flat Tensile Mode. These depend obviously on the load conditions (R, S_{max}) and are influenced also by crack length and frequency. The Tensile Mode appeared dominant at low frequency and in regime of Short Cracks, while the Shear Mode did likewise at high frequency and in the regime of Long Cracks (Fig. 20). The transition from one mode to another depends upon crack lenght and applied load: this may lead to the definition of a variable parameter of the stress state as, for instance, K_{max}, ΔK or ΔK_{eff}. In fact, beyond an opportune critical value of ΔK_{eff}, fracture due to Tensile Mode is no longer present, while at lower values some mixed

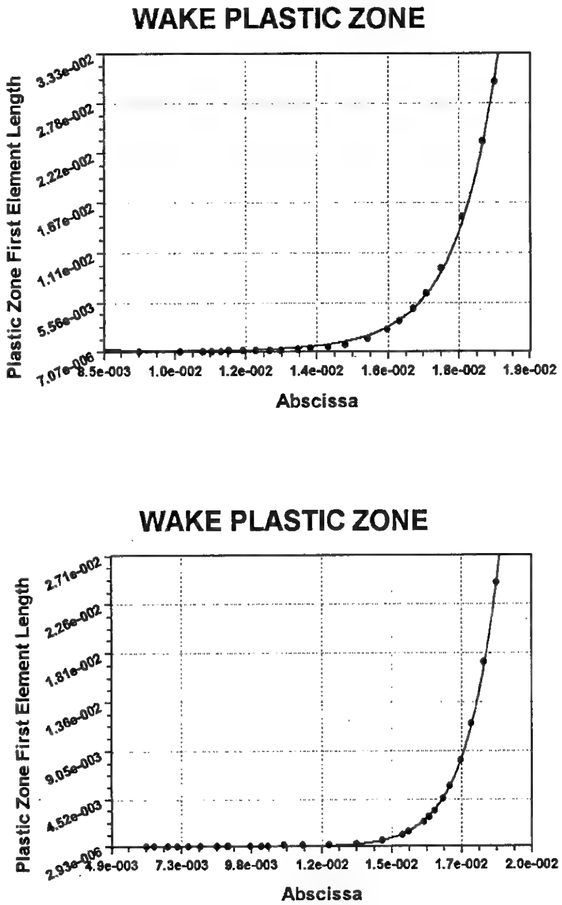

Figure 19: Wake plastic zone spreading for 2195 alloy

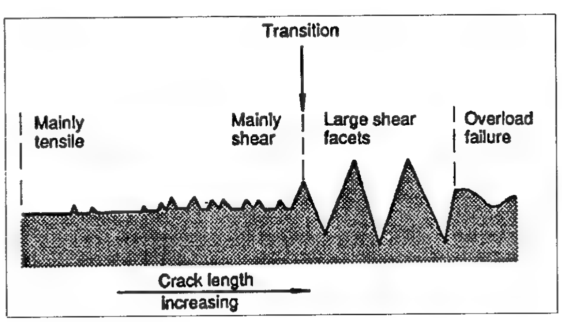

Figure 20: Fracture profile for 8090 alloy

ways of fracture may be observed (Shear + Tensile). Such marked dependence from the effective value of stress intensity factor, has been justified correlating the real dimension of the plastic zone (ρ) with the limit of diffusion of hydrogen per cycle. In other words, when ρ overcomes the distance of diffusion of hydrogen, a fracture is had preferentially by Shear type rather than Tensile type. Basing on this concept, for low values of the ΔK at 1 Hz (Photo 1), the surface shows mixed-mode fracture (Shear + Tensile), that tends, for higher values of the ΔK (Photo 2), to a Shear Mode fracture with the presence of delamination. Applying the chemical solution previously described, the fracture surface revealed (Photo 3) pits which denoted that the crystallografics planes of fracture are (100) type. At higher frequencies (Photo 4), fracture mode is primarily Shear Mode, with an accented roughness of the surface and therefore a notable effect of crack closure. The analysis, performed by Scanning Electron Microscope, for 2195 alloy has denoted the presence of overload and delamination. For instance, Photo 5 refers to the zone of threshold in the test at 5 Hz in which the overloads are well recognised. In the same test, fracture surface, at higher ΔK values, shows zones (along the whole specimen) with fatigue striatures (Photo 6, Photo 7). At collapse, the specimens show a ductile structure with dimples (cone and cup). Delamination occurs similarly to 2091 alloy (Photo 8). In the case of 2 Hz tests, 2195 alloy introduces, at threshold, a behaviour analogous to that at 5 Hz, with less evident overload. Furthermore at 4-8 $mm.$ distance from the threshold the fracture is similar to that of 5 Hz test while, at around 10 $mm.$ distance from the threshold, some fatigue striatures, less marked than in the test at higher frequency, appeared. At higher values of ΔK, just before collapse, is noticed the presence of "Aluminum Plaques" (Photo 9).

Finally in the Photos 10-11-12, at high values of the ΔK, it is easily noticeable, for 2091 alloy, a marked delamination. In this alloy, the surface treatment attack has been performed for the 5 Hz test: the evidence of fracture planes (100) has been much easier in proximity of collapse.

In 2195 alloy, the attack has been performed for the test at 5 Hz. In proximity of the threshold, fracture planes of (111) type have been individuated (Photo 13) while, as crack growth rate and ΔK increase, fracture planes of (110) type are observed (Photo 14).

Photo 1: 8090. Fracture surface

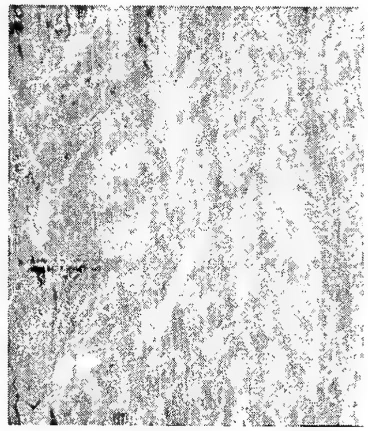

Photo 4: 8090. Roughness and crack closure

Photo 2: 8090. Delamination

Photo 5: 2195. Overloads

Photo 3: 8090. Pits

Photo 6: 2195. Fatigue striatures

Photo 7: 2195. Fatigue striatures

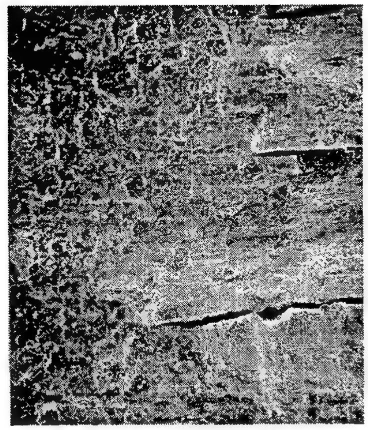

Photo 10: 2091. Delamination

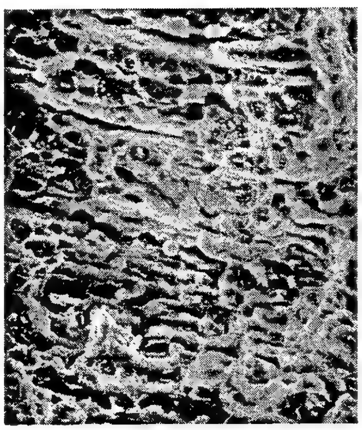

Photo 8: 2195. Delamination

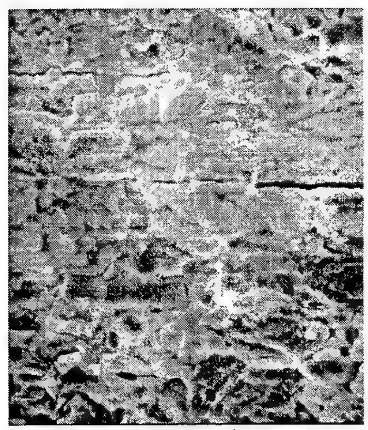

Photo 11: 2091. Delamination

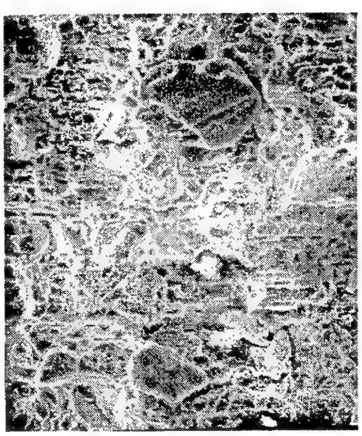

Photo 9: 2195. Aluminium Plaques

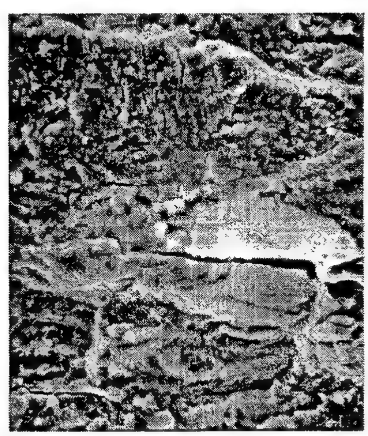

Photo 12: 2091. Delamination

Photo 13: 2195. Fracture planes (111)

Photo 14: 2091. Fracture planes (110)

6. CONCLUSIONS

Fatigue tests on 2091, 2195 and 8090 Al-Li alloys have shown different behaviours in terms of ΔK_{th}, of maximum number of cycles and of plastic crack wake. Average values of ΔK_{th} pass from 3.45 $MPa\sqrt{m}$ for 2091 alloy to 4.40 and 4.58 respectively for 8090 and 2195 alloys, however imposing negligible crack growth for ΔK less then 15 $MPa\sqrt{m}$. While 8090 alloy showed the highest fatigue life at constant frequency in the range 1-10 Hz, 2195 alloy appeared particularly sensible at its variation denoting, as frequency effect, a large gap passing from 1-2 Hz to 5-10 Hz curves. 2195 alloy presented, also, the smallest ammount of plasticity at crack tip, while 2091 alloy showed the widest spreading. Plastic wakes, increasing in size exponentially as crack propagates, appeared to maintain their shape just for the case of 2195 alloy.

SEM crack surface analysis results can be summarized as follows:

- **8090**. Fracture planes of (100) type, evidence of delamination expecially close to failure, no clear striatures and strong effect of crack closure.

- **2195**. Fracture planes of (111) type and of (110) type respectively for low (close to threshold) and high ΔK, strong evidence of delamination close to failure and of fatigue striatures.

- **2091**. Fracture planes of (100) type not easily identificated at low ΔK but subsequently (at failure) much clearer, evidence of delamination and strong oxidation at each test frequency.

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Paper 12

Question by D. Chaumette

The paper did not present on one side analytical methods and on the other side experimental results. Was there a correlation between the two approaches. How was the plastic zone measurement done. What are the effects of plane stress - plane strain.

Author's reply

Experimental tests followed the crack length "a" from which (together with other parameters as W, L, R) the analytical model is able to calculate extension and shape of plastic zone through the use of a series of rigid elements (strip model). This model considers residual plastic deformation along crack surfaces and plastic zone ahead of crack tip.

Small fatigue cracks mostly induce plane-strain conditions. As plastic zone spreads out we assist at a transition to plane-stress conditions. This model, according to Newman, may pass, through the use of a correction factor ß (1-3) from one condition to another depending upon crack size. (5)

A Framework for Corrosion Prediction and Management

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ABSTRACT

Technology and economics drive the reduction in life cycle costs for the sustainment of aging aircraft. Technological advances in metal alloys, heat treatments, coatings and processes provide opportunities for longer component life with less frequent inspections to meet the challenge of maintenance cost reductions. Key aspects for the successful implementation of these newer technologies will be discussed. They include a framework to predict corrosion and the development of a corrosion management paradigm.

INTRODUCTION

Environmental degradation of materials and systems is inevitable over the long term. Therefore, the issue of corrosion is not new; however, a renewed interest in corrosion stems from the aging of our equipment in the budget constrained post-cold war era. The cost of corrosion control for our platforms is staggering. It is beyond the scope of this paper to even estimate this cost, and the interested reader is encouraged to investigate various publicly available documents which detail careful cost estimates.

Structural integrity programs have greatly mitigated the risk of catastrophic failure of components by using a thorough understanding of the mechanics of fracture. Although corrosion can result in the eventual condemnation of individual aircraft over time, the overwhelming concerns are controlling existing corrosion and mitigating the risk associated with catastrophic failure induced or exacerbated by corrosion.

As our existing aircraft are being asked to perform beyond their original design lives or for an extensive number of years, the day to day occurrences of corrosion are resulting in higher maintenance costs.

The focus of this paper is to examine and outline the necessary steps to develop a framework for corrosion prediction and provide suggestions for a corrosion management paradigm. The challenges for corrosion prediction and management are:

- To reduce the cost of ownership for current and future aircraft, and
- To provide cost effective sustainment of aging aircraft.

Many will suggest that the prediction, prevention and management of corrosion are too great and that the challenges are too complex. Our current understanding leads us to conclude that corrosion is fundamentally a stochastic process and the complex mechanism remain largely uncharacterized and unmodeled. Even if the mechanisms of corrosion were well understood, there is little chance of obtaining accurate information on corrosion of fielded components nor would it be possible to offer accurate predictions. The problem with this pattern of thinking, although it is hard to argue with such logic today, is that the consequences are astounding. The implication is that corrosion prevention comes at a very high cost of materials, maintenance man-hours, and capital resources. These costs are incurred through the frequent inspections required to find corrosion and the costly repair or replacement programs required as corrective action when corrosion is found.

Table 1 compares and contrasts structural integrity programs and corrosion. Although the fracture mechanics based structural integrity programs are now quite mature, this has not always been the case. Brought about over time, methodologies have been established. Lifing strategies now accurately manage fracture or durability critical components based on a number of parameters tied back to the cumulative flight hours.

Corrosion is still an immature and largely undeveloped area, not for lack of excellent research but for the lower

Paper prepared for the RTO AVT Workshop on "New Metallic Materials for the Structure of Aging Aircraft", held in Corfu, Greece, 19-20 April 1999, and published in RTO MP-25.

critical components based on a number of parameters tied back to the cumulative flight hours.

Corrosion is still an immature and largely undeveloped area, not for lack of excellent research but for the lower propensity for catastrophic failure of components. Whereas fracture relates to stresses during the flight profile, corrosion is mainly what occurs between flights in local environments that vary widely worldwide. Although corrosion-prone components are well known or easily established, the corrosion critical components analogous to fracture or durability critical components remain largely uncategorized. Presently, there are insufficient methodologies and collected data available to develop a life prediction strategy. The result is that corrosion continues to be a degradation mechanism that is chased and corrected rather than predicted and managed.

CORROSION MANAGEMENT PARADIGM

There are a number of potential avenues for the development of a corrosion management paradigm. This particular method places the burden on the technology community to develop methodologies to enable corrosion management through risk mitigation, emphasizing similarities with structural integrity programs. The following steps give the basis for this paradigm:

- Develop corrosion methodologies,
- Establish known corrodent species and concentrations,
- Establish corrosion critical or corrosion-prone components for tracking,
- Collect and manage data by aircraft tail number
- Establish smart inspections, and
- Provide opportunities for rapid technology insertion.

Develop Corrosion Methodologies

The largest technological undertaking in this management paradigm is to develop detailed scientific understanding of the various types of corrosion including stress corrosion cracking, pitting, intergranular attack, exfoliation, general corrosion and galvanic couples. The priorities for which types of corrosion should receive priority in research has already been carefully studied and reported by the National Research Council.[1] In addition to specific mechanisms for each variety of corrosion, it is important to establish how various corrosion methods interact and what synergism is introduced.

Once sufficient understanding is developed, it will be necessary to codify life prediction algorithms. With limited understanding, these algorithms may simply be empirical relationships representing data trends. However, additional research may work to establish more fundamental relationships and rules akin to an Arrehenius behavior or corrosion analogue to Miner's Rule.

Establish Known Corrodents

Although it is useful to have standardized corrodents and concentrations, such as sodium chloride or synthetic seawater solutions, for comparing new or improved protective coatings or alloys, these fluids may not be ideally suited for verification of corrosion methodologies. Instead, we can go beyond relying on salt fog or other standardized tests that may be overly conservative for specific applications.

A wealth of data can be gleaned from our existing aircraft fleets. Corrosion-prone areas of aircraft will need to be sampled for corrosion. Specific corrodent species and corrosion products can be obtained during these surveys and, using chemical electrophoresis or other techniques, the active corrodent species and concentrations can be established.

Once corrodent species and concentrations are established, appropriate corrosive mixtures can be formulated and used to verify corrosion mechanisms and methodologies, when developed.

Establish Corrosion Critical Components

Inspection records or discussions with aircraft maintainers may easily identify a number of corrosion-prone components. Such components, with historically known corrosion problems, may be the source of frequently recurring replacement and/or inspection. The original equipment manufacturer may also be aware of likely corrosion-prone components based on the use of alloys or heat treatments that may be more susceptible to stress corrosion cracking, or other types of corrosion. Lastly, there may be severe service environments that may be more prone to frequent corrosive damage. These will depend on the type of aircraft; however, they may include areas near lavatories and galleys, around doors, and in wheel wells.

Although the above list represents corrosion-prone components, these should be distinguished from the more important subset of components that are corrosion critical. A corrosion critical component may be defined as one that is difficult to inspect or repair which can deteriorate over time and be a source of significant systems or vehicle failure. It may also include components that, through corrosion, may result in failure which would be mission limiting.

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[1] National Research Council's National Materials Advisory Board, "Aging of U.S. Air Force Aircraft," NMAB-488-2, 1997.

Collect and Manage Aircraft Data

Collecting sufficient amounts of accurate data is the key to both corrosion prediction and prevention. This data will need to be managed by aircraft tail number to accommodate the specific environments and other data required. Key variable will be established from the corrosion methodology, but will include local service conditions such as temperature, humidity and other weather conditions prevailing at the base or airfield where deployed. To a first approximation, if this data is not available, an index system could be employed, ranking various bases and airfields by their propensity for specific environmental factors leading to corrosion. Aircraft maintenance history will also need to be collected and managed, even to incorporate aircraft washings and the use of corrosion preventive compounds.

It is important that this step not become a logistical burden to maintenance personnel. To the extent feasible, this data should be collected automatically. If possible, smart components should be used. It may be possible to have sensors or data loggers attached or embedded for convenience. Whatever the approach, steps should be taken to minimize the need for data entry to avoid the introduction of errors into the database and to keep costs low.

Establish Smart Inspections

To keep the costs of ownership in balance, it is imperative that we avoid:

- Establishing rigid inspection intervals where unnecessary. A first look at this approach, it seems quite obvious that planes in relatively benign corrosive environments may benefit provide cost savings by having broader inspection intervals.
- De-painting for inspection. As better NDI techniques are developed and as corrosion prediction methodologies improve, it will be best to let protective coatings, primers, and topcoats provide the corrosion protection without interruption for unnecessary inspection.
- Reverting to finding and fixing corrosion. Although there is a great deal of momentum to continue with finding and fixing corrosion, the high cost of inspection and premature repair is not the long-term, cost effective solution to the corrosion problem.

By avoiding the pitfalls listed above, it will be easier to develop and implement smart inspection strategies. These will include the use of predicted corrosion rates methodology to identify critical inspections of corrosion critical components. These will not be generically applied to all aircraft based on time in service, but will take into account the service environment for the specific aircraft, by tail number. The long term smart inspection strategy will be to use sensor data to trigger inspection or, better yet, component replacement. Provide Rapid Technology Insertion

Involve the technology community in ways to reduce the cost of ownership for aging and future aircraft platforms. This may entail close coordination with the original equipment manufacturer and supplier base. All due consideration should be given to the practical application of technological advancements. Some of these may include:

- New corrosion resistant alloys,
- Heat treatments for new or existing alloys that may reduce the risk of SCC or other types of corrosion,
- Environmentally-acceptable coatings,
- Faster and more durable repair processes, and
- Non-destructive techniques that can further increase the interval between inspections.

SUMMARY

A complete corrosion management paradigm has been outlined which is based on technology drivers to advance the basic understanding of corrosion mechanisms, and provide a methodology of corrosion to enable corrosion-based life prediction. In addition, fielded aircraft should be the basis for establishing known corrodents and their concentrations for verification of corrosion methodologies developed. The burden falls on the aircraft owners, with the recommended enlistment of the original equipment manufacturer, to identify and establish corrosion critical components for tracking in an extensive database which incorporates the latest in networking and information technology. Inspections are not to be established by the calendar, but by the culmination of the established methodology and accurate data for corrosion critical components. An overarching consideration is the rapid integration of technological advancements that may provide improved component life and be effective for the total cost of ownership.

Table 1: Comparison of Structural Integrity Programs and Corrosion

Structural Integrity	Corrosion
Identifiable fracture critical or durability critical components	Corrosion critical components not identified
Based on cumulative flight hours	Primarily related to non-flight hours
Established methodology	Insufficient methodology established
Fleet Data Tracking	Insufficient data collection and/or management
Predicted and managed	Currently a "find and fix" approach

Advances in Protective Coatings and their Application to Ageing Aircraft

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Abstract

Significant improvements have been achieved in the performance of coatings used in the corrosion protection of military and civil aircraft during the last thirty years. Research into aircraft paints, for example, has resulted in coatings with increased adhesion, fluid resistance and greater flexibility. New methods of paint stripping and novel processes for the repair of pre-treatments and metal coatings are being developed which will lead to reductions in the cost of corrosion maintenance and improved levels of protection. The paper reviews recent developments in aerospace coatings and considers their application in ameliorating some of the corrosion problems associated with ageing aircraft.

1 INTRODUCTION

Advances in aircraft coatings over the last 50 years or so have been in the main part concerned with improving the level of corrosion protection afforded to aerospace components and structures. This has become more critical with the introduction of materials with increased strength and damage tolerance which generally have reduced resistance to corrosion attack. The protection of aluminium parts, for example, has progressed from the use of etch primers over-coated with alkyd and cellulose nitrate/alkyd based paints to schemes involving the use of a pre-treatment, a primer containing a leachable corrosion inhibitor and a polyurethane topcoat [1].

Many of the processes employed in the surface finishing of aerospace parts involve both the use of toxic materials which can be absorbed into the food chain and the use of solvents which can cause damage to the ozone layer and promote smog formation. The protection of aluminium alloys for example is very dependent on the use of chromate-based pre-treatments and chromate pigmented primers. As a result there is increased pressure on the aircraft constructors and operators to adopt environmentally compliant coatings.

This paper first examines the general principles involved in the corrosion protection of aircraft and looks at several examples of in-service corrosion. These have been used to show how the breakdown or failure of protective treatments has led to the initiation of corrosion. The paper reviews the developments in protective treatments and repair methods, which have taken place, and considers how they have been applied to ageing aircraft. Finally new developments in coating technology are discussed and their likely impact on the protection of ageing aircraft is discussed.

2 CORROSION PROTECTION

The protection of both civil and military aircraft is based on reducing the risk of corrosion through design, the selection of materials that are resistant to corrosion and the application of protective treatments to individual components.

2.1 Design

A number of factors need to be considered during the design stage of an aircraft to reduce the risk of corrosion. Some of these are listed in table 1.

Table 1 Corrosion control through design

Potential problem	Design solutions
crevices	use of sealants wet assembly
dissimilar metal contacts	wet assembly metal coatings to reduce differences in galvanic potential shims
water traps	drain holes, fillers
leakage from galleys, toilets	drain paths use of non-metals sealed floor coverings

In some instances, problems experienced on military aircraft have arisen because insufficient consideration has been given to corrosion control through design. The failure on the assembly of components to fill crevices and avoid direct contact between dissimilar metals has often led to the early initiation of corrosion.

2.2 Materials selection

Aluminium alloys easily represent the bulk of the materials used in the construction of ageing aircraft. In many cases high strength aluminium-zinc-magnesium alloys were employed in plate, forging and extrusion product forms in the manufacture of top wing skin panels and landing gear components. Many examples of the 7075 aluminium alloy in the T6 temper can be found on military aircraft presently in service. Similarly 2024 aluminium - copper alloy in the naturally aged T3 temper has been widely used for the construction of lower wing skin panels and fuselage components where good resistance to fatigue cracking is the main requirement. In terms of materials selection for corrosion resistance, 2024-

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T351 and 7075-T651 in the form of plate or other thick section forms are poor choices since both alloys are susceptible to exfoliation corrosion and stress corrosion cracking. Unfortunately this was not evident until aircraft built with these materials had been in service for some years.

In the case of the 7075-T651 plate alloy attempts were made to modify the heat treatment to improve the corrosion resistance whilst accepting some reduction in strength. Duplex ageing treatments were introduced which gave lower susceptibility to exfoliation corrosion and stress corrosion cracking. Table 1 compares various tempers of the 7075 alloy.

Table 2 Corrosion behaviour of some 7000 series alloys

Alloy	Temper	Resistance to stress corrosion cracking	Resistance to exfoliation corrosion
7075	T651	D	D
7075	T7351	A	A
7010	T7651	B	B
7010	T73651	A	A/B
7050	T7651	B	B
7050	T73651	A	A/B

A further development was the production of new aluminium - zinc - magnesium alloys 7010 and 7050 which could be heat treated to the T7651 and T73651 tempers to give resistance to exfoliation corrosion and stress corrosion cracking and strength levels comparable to 7075 in the T651 peak aged condition.

Whilst the use of alloys such as 7075 in the T6 condition should be avoided on future aircraft, the replacement of components on existing aircraft with less corrosion susceptible alloys is generally not an option which has been taken. The approach adopted has been to carry out careful audits of the materials used on various ageing aircraft in order to identify areas which may be particularly prone to corrosion attack. This has been complemented with teardowns of aircraft to look for evidence of corrosion in critical areas. Section 3 gives some examples of in-service corrosion problems. The control of corrosion on parts and areas constructed using susceptible materials has been largely through the use of coatings and surface treatments.

2.3 Protective treatments

The standard protective treatments applied to aerospace components are summarised in table 3.

Table 3 Standard protective treatments for aerospace components

Material	Protective treatments
Aluminium alloys	pre-treatment + chromate pigmented primer
Steels	cadmium plating + passivation
Magnesium alloys	pre-treatment + resin + primer
Titanium alloys	anodising

The main protection applied to aluminium alloys is epoxy primer paint pigmented with strontium chromate corrosion inhibitor. This is applied to the aluminium alloy component, which is pre-treated either by anodising in chromic acid, or by applying a chromate conversion coating. The pre-treatment promotes adhesion of the paint to substrate and additionally provides some measure of corrosion protection.

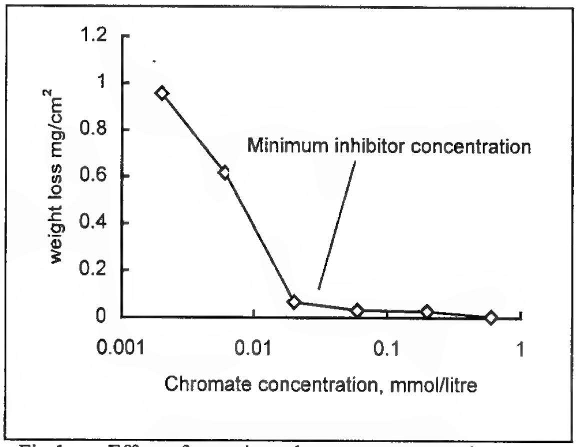

Fig 1. Effect of strontium chromate concentration on the corrosion of 2014-T6 aluminium alloy in 600mM/litre sodium chloride solution

Chromates are very efficient corrosion inhibitors for aluminium alloys particularly in the presence of chloride ions. This is illustrated in fig.1, which shows the effect of strontium chromate concentration on the corrosion of 2014-T4 sheet aluminium alloy in 600mM/litre sodium chloride solution [2]. The data show that provided the concentration is greater than 0.02mM/litre the corrosion attack is prevented. The protection of aluminium alloys by inhibited primers depends on the release of chromate when there is contact with moisture. Chromate based primers currently applied to military aircraft have leaching rates, which ensure that the level of chromate level is well in excess of the critical level given in fig.1. Corrosion is prevented at areas where the paint has become damaged or there is loss of adhesion between the paint and the substrate. Studies made some years ago by Kohler and Scott[3] indicated that chromate levels present in bilge fluid samples were normally well above the critical level indicated in fig.1 to inhibit corrosion.

The preferred scheme for the protection of steel components and fasteners is cadmium plating deposited either by electro-deposition or physical vapour deposition. The cadmium plating acts both as a barrier coating, separating the steel substrate from the environment, and as a sacrificial coating which is able to give protection when the coating becomes damaged. A chromate passivation treatment is applied to the cadmium plating to promote paint adhesion and to improve the corrosion resistance of the coating.

The corrosion protection of magnesium alloys is achieved by applying a barrier coating. The current trend is away from the use of magnesium alloys on aircraft mainly because of its poor corrosion resistance. However many of the ageing military aircraft in service have employed magnesium alloy castings

for gearbox housings, undercarriage components and canopy structures. Magnesium alloy sheet has also been employed for skinning on helicopters and on the rear fuselages of fixed wing aircraft. Severe galvanic corrosion problems arising from cracking of the protective coating around fasteners resulted in major repair programmes with the replacement of magnesium sheet with aluminium alloy sheet. Current design documentation prohibits the use of magnesium alloy sheet on UK military aircraft.

Titanium alloys are generally resistant to corrosion when exposed to aircraft fluids and marine environments. Apart from cleaning they require no further treatment although components are often painted. The normal practice would be to pre-treat by wet abrasive cleaning, etch priming, pickling or anodising and then paint with an epoxy primer. When titanium alloy parts are in contact with components machined from magnesium alloys or aluminium alloys coatings are applied to reduce the risk of galvanic corrosion. Zinc based and aluminium based coatings are frequently employed for this purpose. Coatings are also applied to titanium alloy parts to improve their wear and fretting resistance.

3 IN-SERVICE CORROSION

The various types of corrosion found on fixed wing aircraft and helicopters have been well documented in the AGARD corrosion handbook [4]. A few examples are given below of some of the more common corrosion problems that have been observed by the authors on ageing transport aircraft and fast jets.

These include: -

1. corrosion problems associated with materials selection i.e. use of materials which are inherently susceptible to various forms of attack such as intergranular corrosion, exfoliation corrosion and stress corrosion cracking
2. Problems arising from poor design considerations - e.g. crevice corrosion and dissimilar metal corrosion.

Exfoliation corrosion has been found on the lower wing skins of several military aircraft types currently in service in the United Kingdom. The problem has arisen with panels machined from 2024-T351 plate material. This material has a poor resistance to intergranular and exfoliation corrosion but was originally selected for lower wing skin applications because of its excellent fatigue properties. In each instance breakdown of the protective treatments applied when the aircraft was built, has allowed the metal substrate to become exposed to the environment. Two examples of exfoliation corrosion occurring in-service are described below.

In the first example, extensive exfoliation corrosion was found to have taken place on the internal surfaces of the lower wing skin of a transport aircraft. The main cause of the problem was the leakage of hydraulic fluid. This had degraded the protective scheme, effectively acting as a paint remover. The aircraft was operated in a marine environment and the build up of moisture and led to the initiation of intergranular attack eventually leading to the development of exfoliation corrosion. The protective treatment which had been applied when the aircraft was first built consisted of chromic acid anodising, a prime coat of etch primer and an epoxy top coat pigmented with aluminium. During subsequent repair, the protective scheme was over-painted with a gloss polyurethane finish to give improved resistance to hydraulic fluids.

In the second example, exfoliation corrosion was found to have developed on the external wing surfaces of a fast jet at areas adjacent to fasteners. The attack initiated in the countersinks and spread parallel to the wing surface. Two factors were believed to play an important role in the initiation of corrosion. The first was the absence of sealant between the fastener and countersink and the second was the cracking of the polyurethane finish. As a result a crevice existed in the fastener \ countersink area permitting the ingress of moisture and the development of exfoliation corrosion. Subsequent repairs involved the blending out of corrosion damage and re-protection using coatings with improved flexibility to reduce the risk of cracking.

Many of the instances of stress corrosion cracking (SCC) found on ageing aircraft occur in parts machined from thick section 7000 series alloys heat-treated to the T6 peak aged condition. Examples of SCC found recently on UK military aircraft include: -

1. SCC in main spars manufactured from material equivalent to 7075-T6. Crack initiation was associated with deterioration of the protective scheme around fasteners
2. SCC in extrusions used for front and rear spar booms made from 7000 series alloys
3. SCC in undercarriage components machined from 7000 series alloy forgings

An example of galvanic or dissimilar metal corrosion was identified on flap shroud panels manufactured from 2024-T351 aluminium alloy. At areas where titanium alloy flap tracks were fastened to the flap shroud panel there was enhanced corrosion attack as result of dissimilar metal contact. The problem was thought to be associated with the absence of wet assembly compound between the titanium track and aluminium alloy flap shroud.

An example of crevice corrosion was found on a leading edge slat that had been manufactured from a clad 2014-T6 aluminium alloy. A modification had been carried out using a stainless steel stiffener, which was adhesively bonded to the slat to improve its fatigue performance. Extensive corrosion was found to have taken place on the clad aluminium alloy skin where it was covered by the stainless steel stiffener. The attack was intergranular in nature and in places penetrated to a depth of 50% of the skin. The stainless steel stiffener had been cadmium plated to reduce the risk of dissimilar metal corrosion with the aluminium alloy skin. Examination of the contacting surfaces indicated that there was only a patchy layer of sealant or adhesive present. It was concluded that corrosion had initiated as a result of the adhesive bond breakdown. The establishment of a crevice would further lead to accelerated corrosion of the aluminium alloy slat.

Invariably problems have arisen with materials, which are inherently susceptible to corrosion as discussed in section 2.2. The cause is associated with a breakdown of the protective scheme, which has been applied. Internal schemes for example are applied when the aircraft is constructed and are generally expected to last the life of the aircraft. Apart from

the application of a supplementary protective scheme to enhance the existing protective treatment, there is usually little opportunity to change or improve the internal protection.

4 RECENT DEVELOPMENTS IN PROTECTIVE COATINGS

4.1 Etch primers

One of the early paint schemes applied to military aircraft was an etch primer, top coated with an epoxy finish. This scheme has not been employed for a number of years and etch primers now only find applications as pre-treatments over which is applied a standard epoxy primer. One of the main developments has been the formulation of etch primers which give enhanced filiform corrosion resistance. BS 2X32 [5] is the current UK standard for etch primers and includes a filiform corrosion test.

A recent application of etch primers has been in the re-protection of the air intake of an aircraft following paint stripping and removal of the corrosion damage. An etch primer was chosen as the pre-treatment rather than the normal chromate conversion coating. The choice was made to avoid the need to deoxidise the surface and water rinse the intake before and after the application of the conversion coating. There was concern that aggressive solutions could have been washed into the internal structure of the aircraft.

4.2 Paints

4.2.1 Epoxy primers

For a number of years, chromate pigmented epoxy primers specified for use on UK military aircraft were qualified to MoD specification DTD 5567 [6]. Developments in the mid 1980s were concerned with improving the adhesion and fluid resistance of the standard primer. This was in response to corrosion problems occurring on civil transport aircraft. Leakage of hydraulic fluids led to degradation of the standard protective coatings and the eventual corrosion attack of the underlying aluminium alloy substrate. New generation primers were developed by the leading aircraft manufacturers, which in addition to giving improved resistance to aircraft fluids also gave greater adhesion.

The current UK aerospace specification for two component epoxy primers is BS 2X33[7] and covers two material types. Type A materials are intended for application to chemically pre-treated substrates suitable for general applications and are equivalent to materials qualified to DTD5567. Type B materials have improved tolerance to the standard of surface preparation and increased chemical resistance. This is reflected in differences in the condition of the substrates used in the qualification tests for paint adhesion. For type A materials, cross hatch adhesion measurements are conducted on aluminium alloy panels (BS L163) which have been acid chromate pickled prior to priming. For type B materials on the other hand measurements are carried out on detergent degreased panels. In each case the pass requirement is the same. For resistance to hydraulic fluid tests the pass criteria is more severe for the type B materials. For example after immersion in tri-*n*-butylphosphate, the load applied in the scratch test for type B materials is 2000g compared with 1000g for type A materials.

For most applications on military aircraft the type B materials are now used.

4.2.2 Acrylic finishes

Acrylic finishes have been used on military aircraft in the United Kingdom over a number of years. They were introduced in the 1970s partly to fulfil a requirement for a paint finish, which could be more readily removed than polyurethane coatings. The UK MoD specification DTD5599 [8] covers this finish. Hoey [9] has indicated that the useful life of an acrylic finish is 2-3 years compared with 5 years for a polyurethane finish. For some time the trend has been away from acrylic finishes and polyurethane schemes with their higher resistance to fluids are now preferred. A further disadvantage of acrylic finishes is their relatively poor resistance to chemical agents. Most fixed wing aircraft are finished in polyurethane schemes and helicopters are being finished in polyurethane schemes when they are repainted.

4.2.3 Polyurethane finishes

Polyurethane finishes have been available for aerospace applications for a number of years. The MoD specification DTD5580 [10] describes the original scheme applied to UK military aircraft. As highlighted earlier in this paper (section 3) one of the main sources of corrosion initiation has been identified as cracking of the paint film around fasteners. One of the main developments in polyurethane finishes has been the formulation of paints, which give coatings with increased tolerance to flexing, a particular problem with large aircraft and helicopters. The current specification for polyurethane finishes is BS 2X34[11]. The specification covers two types of materials type A and type B. Type A materials are intended for interior and exterior use where maximum resistance to fluid attack is required whilst type B materials are intended for exterior surfaces, and offer increased tolerance to flexing compared with type A materials.

4.2.4 Selectively removable paint schemes

The removal of paint from the exterior surface of an aircraft is manpower intensive and involves the handling and disposal of hazardous materials. One approach has been to develop a selectively removable paint scheme which permits the polyurethane top coat to be removed leaving the primer intact. The scheme uses an intermediate coat, typically 8-12μm thick, which is applied over a standard epoxy primer. This is then top coated with a polyurethane finish. Blackford [12] has described the use of such schemes on Concorde and Airbus A320 aircraft. Trials have been conducted demonstrating that the polyurethane finish may be removed used a comparatively simple paint stripper.

The recently issued specification BS X35[13] describes the requirements for a selectively removable intermediate coating for aerospace purposes. Two types of intermediate coating are specified. Type A intermediate coating is intended to function with a finish conforming to type A of BS X34 where maximum resistance to fluid attack is required. Type B intermediate coating is intended to function with a finish conforming to type B of BS X34 offering increased tolerance to flexing compared with type A finish. A specification covering the paint remover (BS X36 [14]) has also been issued. A composition for a reference paint remover based on benzyl alcohol is given which meets new VOC requirements.

4.2.5 Low VOC materials

Legislation concerning the release of solvents into the atmosphere was introduced under the 1990 environmental protection act. One area to have an impact on the aerospace industry was on the permitted level of volatile organic compounds in aircraft paints. The current limits are summarised in table 4 and are based on figures given in reference 15.

Table 4 Volatile organic compound emission concentration limits

Coating	VOC emission concentration limits g/litre
Pre-treatment primer	780
Primers	350
Selectively removable intermediate coats	780
Paint removers	300
Top coats	420

The present policy is to procure materials which meet the VOC levels given in table 4 whilst complying to the performance requirements given in the appropriate paint specification. For example the epoxy primer currently being applied to military aircraft is a high solids material. It meets the requirements of BS 2X33 and has a VOC level of less than 350g/litre. This compares with a VOC level of 600 g/litre in the original primers. At present only gloss finishes are available which meet the VOC requirement in table 4. Matt finishes which are used on most military aircraft are still under review.

4.3 Elastomeric coatings

A number of corrosion problems arising on ageing aircraft are related to the cracking of the protective finish around fasteners and the subsequent ingress of moisture. For this reason the BS X34 type B topcoat with greater flexing properties is preferred to the type A finish. For some areas however this finish alone does not provide adequate protection and schemes using elastomeric coatings have been employed. These coatings are self-curing polysulphide based systems which are either applied directly as a primer or are used over a conventional epoxy primer. They have been employed both in the repair of corrosion damage on exterior surfaces and in interior areas where moisture collects and conventional schemes cannot be applied. An example of the latter was in the re-protection of an area housing a fuel bag tank. Moisture collected under the tank causing breakdown of the protective treatment and the eventual development of corrosion. After blending out any corrosion present the surface was re-protected using and etch primer and a polysulphide coating.

One disadvantage of the elastomeric coatings is their thickness. Typically they are applied at a thickness of between 150 to 250μm giving coating weights of around 8oz/sq yard compared with 2oz/sq yard for conventional paint films. Their use has been limited to exterior areas on upper and lower wing skins where corrosion has occurred adjacent to fasteners. One scheme used has replaced the standard epoxy primer with a polysulphide coating which is then over-coated with a flexible polyurethane coating. A second approach has been to apply the elastomeric coating over a conventional epoxy primer but to restrict it to fastener runs. The surface is then topcoated with a flexible polyurethane finish. Cracking can still occur however due to the mismatch of flexing properties between the different coats. The aim is to improve the corrosion protection at fasteners and minimise the weight penalty.

4.4 Metal coatings

Three areas of coating development have made a significant impact on the corrosion protection of military and civil aircraft. The initial thrust has been in the development of replacement coatings for cadmium and chromium plating but several applications have been identified where they have been used to achieve improved corrosion protection. Developments in electrodeposited zinc alloy coatings, aluminium coatings and metallic-ceramic coatings are discussed below.

4.4.1 Electrodeposited zinc alloy coatings

Electrodeposited zinc alloy coatings have been considered as alternatives to cadmium plating for the protection of steel parts and fasteners. The main interest has been in electrodeposited zinc-nickel and zinc cobalt coatings.

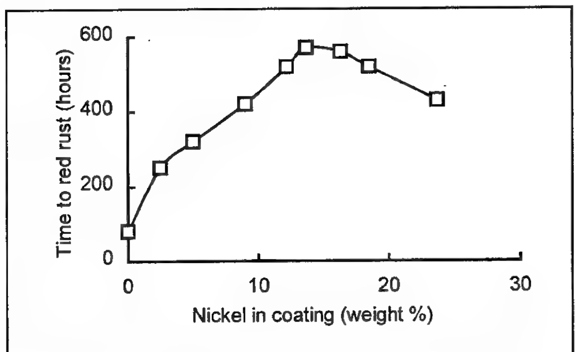

Fig. 2 Effect of nickel concentration on time to first rust on exposure to neutral salt fog

The addition of nickel to zinc coatings greatly improves the corrosion resistance on exposure to neutral salt fog. Fig.2 shows the effect of nickel concentration on the time to first rust [16]. The optimum composition is achieved at ~14% nickel which represents the balance between the barrier and sacrificial properties of the coatings.

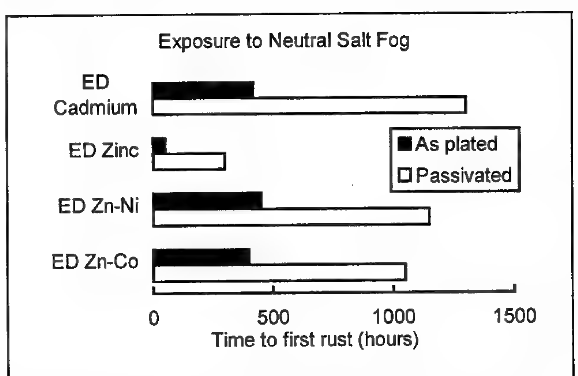

Fig.3 Corrosion behaviour of steel panels with 8μm thick coatings.

Both acid and alkaline zinc - nickel plating baths are available commercially and have been examined as possible alternatives to cadmium plating.

Fig.3 compares the corrosion of several zinc alloy coatings exposed to neutral salt fog [17]. The results obtained show that under these conditions levels of corrosion protection similar to cadmium may be achieved. Currently some aerospace parts manufactured from medium strength steels are being electroplated with zinc alloy coatings.

4.4.2 Aluminium coatings

Aluminium coatings prepared by physical vapour deposition have been generally available in the aerospace industry for more than 20 years. There applications include the protection of steel parts and fasteners as an alternative to cadmium plating and the plating of titanium alloy fasteners to prevent dissimilar metal corrosion in contact with aluminium alloys. Within the UK PVD aluminium coatings have found comparatively applications on military aircraft. One area where these coatings have been considered is in the protection of parts manufactured from 7075-T651 aluminium alloy where stress corrosion cracking is a concern. Initial laboratory studies [18] have indicated that PVD aluminium coatings may help in delaying the initiation of stress corrosion cracking.

4.4.3 Metallic-ceramic coatings

A variety of commercial coatings are currently available which may be described generically as metallic - ceramic coatings. These consist of an inorganic matrix containing a dispersion of metal flakes or powders. For corrosion protection purposes the most interesting systems are those that are zinc or aluminium based. The coatings have been considered as cadmium replacements however to date their use for this purpose has been fairly limited. Mosser [19] has described the use of aluminium-ceramic coatings as alternatives to cadmium plating for some undercarriage applications.

5 IMPROVED REPAIR SCHEMES

A recent estimate made for the RAF Tornado fleet indicate that on average 296h are spent on each aircraft annually on the removal of corrosion and repair of the protective schemes and equates to an annual cost of around £2M [20]. Calculations have shown that even a modest reduction in corrosion arisings could have a significant impact on aircraft availability. For instance if the number of corrosion arising should be reduced by 10%, the time spent by the Tornado fleet on 3rd line maintenance would be reduced by 56 days. Aircraft availability can be further improved by introducing more effective methods of repair, which reduce the time spent on repair and give improved protection. Three areas are considered below, paint removal, corrosion removal and the use of brush plating and anodising techniques.

5.1 Paint removal

The paint schemes applied to the external surfaces of aircraft generally have a service life of between 4 and 5 years. The normal practice has been to remove all the protective treatment from the aircraft surface using a chemical paint stripper to expose the metal substrate. This allows inspection of the airframe for evidence of corrosion attack and fatigue cracking.

Most military aircraft are finished with a polyurethane topcoat selected because of its high resistance to fluids. The only chemical paint strippers, which are effective on these types of coating, are based on methylene chloride with additives of phenol. These are unpleasant and hazardous to use and much effort has been directed towards finding more effective and safer methods of paint removal.

Plastic media stripping (PMS) has become an important method of removing paint from military aircraft. The process involves the impacting of small plastic beads onto the painted surface. The equipment developed allows the plastic beads to be blasted at the work surface at a controlled velocity and to recycle spent beads removing dust and paint particles. Facilities have been installed allowing small components to be stripped as well as complete aircraft. There are still some concerns regarding the treatment of thin sheet, clad alloys and composite materials. Other blasting media such as dry ice, and wheat starch have been evaluated. There is also interest in laser techniques, possibly for use at areas around fasteners.

5.2 Corrosion removal

Impact blasting techniques such as shot peening and abrasive blast cleaning are widely used in the aerospace industry for the preparation of metal surfaces. In shot peening operations, the workpiece is blasted with a high velocity stream of spherical particles such as glass balls or steel shot in order to induce compressive stresses into the surface. The process is mainly used to improve the fatigue strength of components and to give improved resistance to stress corrosion cracking. Abrasive blast cleaning is used to remove corrosion products or scale from metal components or to roughen the surface in preparation for bonding, painting or metal coating. The abrasives which are employed include alumina grit, angular metallic particles, crushed slag and smooth glass beads.

In recent years abrasive blast cleaning with either small diameter glass beads or fine alumina grit has been used to blend out corrosion damage on aircraft. One area where it has proved to be invaluable has been in the repair of corrosion damage that sometimes occurs adjacent to countersink fasteners, particularly on upper wing skins. Prior to the introduction of abrasive blast cleaning, the corrosion was blended out by hand using metal wool, abrasive pads or small abrasive wheels mounted in a power drill. This could take up to 2 hours for a single fastener head but by using abrasive blasting techniques this time has been reduced to a few minutes.

When abrasive blast cleaning methods were first introduced for use on military aircraft, glass beads were preferred to alumina grit because they readily remove brittle corrosion products but remove little of the ductile metal substrate. It became apparent, however, that although the surface appeared free of corrosion after blasting the peening action of the glass beads could deform the surface layers and cause small pockets of corrosion in pits or intergranular sites to become trapped. There was concern that these pockets of buried corrosion might act as stress concentrators which could accelerate fatigue crack initiation or allow enhanced corrosion attack. A change to alumina grit blasting was therefore made in order to ensure that all the corrosion was blended out even though this would lead to a significant amount of metal removal.

Research conducted at DERA by Smith and Hewins [21] examined the effects of abrasive blast cleaning on the fatigue of a 2014-T6 aluminium alloy. Table 5 compares the fatigue strength at 10^7 cycles determined using rotating bending tests. The results indicate that although there is a significant loss in fatigue strength following exposure to neutral salt fog this can largely be restored by abrasive blasting.

Table 5 Fatigue strength (M Pa) at 10^7 cycles

	None	8h exposure to neutral salt fog
As machined	170	107
Alumina grit blasted	165	150
Glass bead blasted	175	160

Alumina grit blasting is now extensively used in the blending out of corrosion damage.

5.3 Brush plating

Brush plating techniques allow the in-situ repair of metal coatings on aircraft. The main application has been on the repair of damaged cadmium plating on landing gear components. Research undertaken at DERA has explored the use of brush plated zinc alloy coatings for the repair of corroded metal coatings [22]. Trials were conducted on a commercial zinc-nickel coating and on two experimental coatings, a zinc-nickel and a zinc-cobalt. In neutral salt fog tests, encouraging results were obtained with the experimental zinc-cobalt system. A series of experiments were conducted using steel test panels which had been plated with electrodeposited cadmium, PVD aluminium, electrodeposited zinc-nickel or electrodeposited zinc-cobalt coatings. The panels were then damaged by removing the centre portion of the coating. The bare steel was then repaired by brush plating. Corrosion tests indicated that brush plated zinc - nickel and zinc - cobalt coatings could be used to repair a range of coatings including bath plated zinc-nickel, zinc-cobalt, PVD aluminium and bath plated cadmium. Further details of the research programme are given in reference 22.

5.4 Brush anodising

Recent research at DERA has examined the use of brush anodising techniques for the repair of protective treatments [23]. Trials were conducted on two sheet aluminium alloys, an aluminium copper alloy 2014-T6 and an aluminium - zinc - magnesium alloy 7075-T6. Three commercial brush anodising treatments were evaluated, a sulphuric acid solution, a boric-sulphuric acid solution and a chromic acid control. Corrosion tests were made on samples, which had been bath anodised and then damaged and repaired using brush anodising. The results obtained showed that satisfactory repairs could be made using the boric-sulphuric acid brush anodising process. The work is at an early stage and further research is needed to develop a practical repair process.

6 FUTURE COATING DEVELOPMENTS

Many of the developments in coating technology for aerospace applications are being driven by the need to replace existing protective treatments with environmentally compliant systems.

Trials have been conducted at DERA on a number of commercial chromate-free primers [24]. Accelerated and marine exposure trials have shown that these coatings do not produce the same level of protection as the standard chromate pigmented epoxy primers currently in-service. Under laboratory tests, the chromate-free systems failed to give protection to the substrate at areas where the paint scheme was damaged. This was also observed in marine exposure trials and additionally evidence of filiform corrosion was found after 2 years with a number of the schemes. It was concluded that the most promising materials could have applications on the exterior surfaces of civil aircraft operating in a relatively benign environment but were unsuitable for use on military aircraft.

Research is currently in progress within Europe to develop new surface cleaning and etching processes, which reduce the use of solvents and chromates. Replacements for chromate based anodising processes and conversion coatings are also being investigated and attempts are being made to develop chromate-free primers and sealants. The eventual aim is to introduce a chromate-free protection scheme that will be used both in the manufacture of new aircraft and in the maintenance and repair of aircraft in service.

The last 30 years has seen the development of advanced tape technologies for aerospace applications. This has included protection against erosion, corrosion, abrasion and impact damage. One area currently being evaluated is the use of appliqué technology for the replacement of aircraft topcoats. This could have a major impact on the protection and maintenance of ageing aircraft by reducing the requirement for extensive painting and paint removal facilities and increasing aircraft availability.

In the field of metal coatings, some of the major advances relevant to aerospace have been in the field of PVD coatings. Coatings under development include:-

- aluminium alloy coatings to replace cadmium plating on steel components [25]
- multilayered coatings for fasteners and parts where combinations of corrosion and wear resistance are needed

Although initially intended for new aircraft it is likely that these coatings will ultimately find applications on ageing aircraft to improve corrosion protection.

7 CONCLUSIONS

The protection of military aircraft against corrosion is based on the selection of corrosion resistant materials, the application of protective treatments and careful design to avoid potential corrosion problems such as crevices, dissimilar metal contacts and water traps. One of the main problems associated with ageing aircraft is the extensive use of structural materials that are inherently susceptible to corrosion. In many instances the protective treatments applied when the aircraft was built have proved inadequate and poor design against corrosion has resulted in many corrosion arisings. Although there have been advances in aluminium alloy technology with the development of new tempers and

compositions which combine good mechanical properties with high corrosion resistance, material substitution is generally not an option. Instead the approach has been to look for more effective repair methods and coatings. In general developments in coatings have been incremental rather than major leaps forward. Paints with increased fluid resistance, flexibility and adhesion are replacing existing schemes. An added problem facing the maintenance and repair of ageing aircraft is environmental legislation. This is having a major impact on the aerospace industry to the effect that many of the current protective schemes will not be available in the future.

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Structural integrity
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Fatigue (materials)
Corrosion
Benefit cost analysis
Stress analysis
Protective coatings
Durability
Aging aircraft

14. Abstract

This workshop dealt with the replacement of Structural components of aging aircraft with components manufactured from materials with specifications of a high qualification, thus enhancing various parameters including overall life cycle cost technology (LCC).

The following topics were treated:

- An Overview
- Aluminium Alloys and Composites
- Processing, Fatigue and Durability

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